ADVANCED MISSION CONCEPTS FOR OUTER PLANETS EXPLORATION

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by

Science Applications International Corporation
1701 East Woodfield Road
Schaumburg, Illinois 60195

for

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FOREWORD

This study was conducted between December 1984 and July 1985 as part of the work performed by Science Applications International Corporation under Task 5 of Contract No. NASW-3622, Advanced Mission and Information Studies, for the Solar System Exploration Division, Code EL, NASA Headquarters. The results are intended to assist NASA planners in assessing the requirements, capabilities and programmatic issues associated with mission concepts for the advanced exploration of the outer planets and their satellites.

Stephen J. Hoffman served as study leader for this effort. Significant contributions to this study were made by SAIC staff members including Kevin Cole, Harvey Feingold, Alan Friedlander, Peggy Hastings, Deanna Limperes, Terri Ramlose, John Soldner and Dan Spadoni.

EXECUTIVE SUMMARY

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INTRODUCTION

The outer planets, from Jupiter with its diverse assemblage of 16 satellites to the virtual double planet of Pluto and Charon, exhibit a wide range of natural phenomena which differ greatly from anything found on Earth or among the inner planets. This variety has kept the science community active over the past several centuries attempting to determine the nature of these bodies, and comparing them to one another to better understand their differences and their similarities. The comparatively recent introduction of interplanetary spacecraft to this investigative process has done much to reveal the nature of these planets but has also generated an entirely new set of queries as each world is examined in greater detail.

Given the situation described above, the purpose of this study is to develop a "broad brush" picture of advanced mission concepts for outer planet exploration in the 1995-2015 time frame. This set of missions could satisfy most, if not all, of the Space Science Board recommendations for the outer planets (discussed in detail later in this report) not met by the proposed Solar System Exploration Committee (SSEC) Core Program. Each mission has a sufficiently detailed description of science objectives to allow candidate science instrument payloads, spacecraft functional requirements and mass breakdowns to be defined. This information allows performance and flight time trade studies to be conducted as well as key operational characteristics and design issues to be identified. A cost estimate for each mission is then developed, reflecting both the heritage from Core Program missions and the new technology required to meet the science objectives.

The first three sections of this executive summary briefly describe the assumptions and constraints imposed on the missions by the science objectives, hardware capabilities and available trajectories. Each of 11 candidate missions is then briefly discussed, describing: the specific science objectives to be addressed; hardware elements to be used; trajectory information; and estimated costs.

SCIENCE OBJECTIVES

Beginning with the assumption that the Voyager mission to Uranus and Neptune and the Galileo mission to Jupiter will be successful, the Committee on Planetary and Lunar Exploration (COMPLEX) established a strategy of scientific goals and objectives to be met for any future missions to the outer planets (Ref. 2). In light of the fact that Galileo will address the most important objectives at Jupiter (as established by COMPLEX in 1979), Saturn and its system have been given the highest priority for subsequent missions. The science objectives for Saturn as outlined by COMPLEX are:

1. Intensive study of the Saturn system as a whole;

2. First-order characterization of Titan's surface to guide the planning of future missions;

3. Determination of the composition and structure of Titan's atmosphere; and

4. Determination of the composition and structure of Saturn's atmosphere.

These objectives will be met by the proposed SSEC Core Program missions to Saturn. The first of these, the Saturn Orbiter/Titan Probe (Cassini), will focus on the first three objectives while the Saturn Flyby/Probe will address the fourth.

Further long-term objectives for the outer planets include:

1. Conduct a detailed long-term study of Saturn's rings, small satellites and magnetosphere;

2. Characterize the physical state of Titan's surface;

3. Conduct exploration and intensive study of the Uranus and Neptune systems with special attention to Neptune's moon Triton;

4. Conduct investigative studies on the planetology of the Galilean satellites and Titan;

5. Conduct intensive studies of Jupiter's inner system, including its satellites, inner magnetosphere, and time-dependent phenomena; and

6. Conduct measurements to the base of the cloud layer (or deeper) at the gas planets.

These objectives are the focus of the candidate mission concepts that are presented in this report. Any level of priority for these objectives will be dependent upon the results obtained from Voyager and Galileo.

HARDWARE ELEMENTS

Meeting these science objectives will, in most cases, require only the use of hardware concepts which have been derived or modified from existing elements built and tested for other missions. In this respect, the present study will follow the precedent set by the Mariner Mark II program in that hardware from diverse sources will be assembled to carry out the desired mission. For the mission concepts under consideration, there have been several previous spacecraft with similar flight profiles. For example, the Viking lander, the Voyager bus structure and antenna, and the Pioneer Venus probe pressure vessel, as well as the Galileo engineering subsystems and entry probe represent hardware elements which can be applied to other missions.

Several missions may require hardware which has no analogue in previous spacecraft. In these cases, previous mission and vehicle design studies will be used to generate any required information.

In order to maintain a reasonable level of reliability for these missions, a flight time of less than 10 years has been imposed on all missions. To meet this requirement, two new concepts in the areas of upper stage propulsion and capture method must be invoked. Low-thrust propulsion and aerocapture are thus assumed to be available at the time of launch for any particular mission even though no definite plans to develop either technology exist at present. However, as will be seen in the following sections, both of these technologies will significantly improve the mass performance and flight times to the more distant planets.

The major hardware elements needed to carry out the proposed mission objectives have been listed in Table 1. This table also indicates the heritage assumed for each element and approximate mass values (rounded to the nearest five kilograms).

Table 1

MAJOR HARDWARE ELEMENTS

Hardware Element	Heritage	Estimated Mass (kg)
Support Bus (Ballistic)	Mariner Mark II/ Voyager/Galileo	745
Support Bus (Low-Thrust)	New	1360
Soft Lander	Viking	500
Hard Lander	New	30
Penetrator	Mars Penetrator Concept	55
20 bar Atmospheric Probe (without heat shield)	Galileo	90
100 bar Atmospheric Probe (without heat shield)	Pioneer Venus/Galileo	195
1000 bar Atmospheric Probe (without heat shield)	New	215
Balloon-Supported Probe	New	110
Airship-Supported Probe	New	335
Nuclear Electric Propulsion Stage (Dry)	New	5145
Solar Electric Propulsion Stage (Dry)	New	1200
Aerocapture Vehicle	New	Mission-Dependent

All mass values shown do not include science instruments or propulsion systems and thus represent only the support-type subsystems. In addition, minor variations may be made in each of these values based on the peculiarities of each mission.

The mixture of new and updated hardware technology will be accounted for in the cost estimate for each mission. The SAIC Cost Estimation Model for Advanced Planetary Programs (Ref. 30) used for these estimates accounts for varying degrees of heritage for each of the hardware elements.

PERFORMANCE TRADE STUDIES

Mission performance capability is a function of many parameters, including launch vehicle/upper stage, interplanetary flight mode, trajectory type (which can be a function of launch year), the method of orbit capture, and the type of retropropulsion used for post-launch mission phases. The complete scope of options for each of these categories is summarized in Table 2.

The launch vehicle/upper stage combinations determine the total mass which can be injected into an interplanetary trajectory at the proper energy. The set used for this study includes the Centaur family of vehicles, plus a conceptual propulsion system derived from recent space-based Orbital Transfer Vehicle (OTV) studies at NASA/Marshall Space Flight Center. For this vehicle, designated OTV(4-R)/OTV(2-E), a reusable four-tank OTV (OTV(4-R)) propels the launch stack to a maximum 24-hour orbit, then returns to a Space Station-compatible orbit via aeromaneuvering. A two-tank expendable OTV (OTV(2-E)) then launches the spacecraft to escape from the perigee of the first-stage orbit.

Ballistic, solar electric propulsion (SEP) and nuclear electric propulsion (NEP) flight modes are all considered for the missions presented in this report. The delivery options are further increased by examining both direct and indirect trajectory types, as well as Jupiter gravity-assist swingbys. For missions to the outer planets, trajectories utilizing Jupiter gravity-assist swingbys can greatly reduce launch energy requirements and mission trip times. These trajectory types are especially useful for Uranus and Neptune missions due to the excessive launch energy and flight time requirements for direct ballistic trajectories.

The only trajectory types studied using SEP were the 2+ and 3+ SEEGA. The ballistic equivalent of this Earth gravity-assist trajectory type, the ΔVEGA, was also examined and both SEEGA and ΔVEGA were analyzed employing Jupiter gravity-assists. The advantage of the indirect trajectory types is the ability to capture the mission with a less capable launch vehicle due to the decreased launch energy requirements. However, these trajectory types will, in general, increase the total trip time by the length of time spent on the Earth-to-Earth leg.

The great potential of NEP application to outer planet missions lies in the fact that the nuclear reactor power source operates independently of its distance from the sun. This characteristic of useful thrust acceleration at a large heliocentric distance allows the vehicle to slow down near the target planet and spiral into orbit capture. Likewise, the NEP stage can be used to spiral out from a nuclear-safe orbit about the Earth to escape conditions.

Table 2

PERFORMANCE TRADE OPTIONS

- LAUNCH VEHICLE/UPPER STAGES
  - SHUTTLE/CENTAUR(G)
  - SHUTTLE/CENTAUR(G')
  - ON-ORBIT-ASSEMBLED OR FUELED (OOA) CENTAUR(G')
  - OOA CENTAUR(G')/CENTAUR(G)
  - OTV(4-R)/OTV(2-E)

- INTERPLANETARY FLIGHT MODES
  - BALLISTIC
  - NEP: 100 kW_e, I_{sp} = 5500 sec, η = 0.776, M_{DRY} = 5145 kg
  - SEP: P_A = 32 kw (BOL), P_o = 28 kw (BOL), I_{sp} = 3560 sec, η = 0.682, M_{DRY} = 1200 kg

- TRAJECTORY TYPES
  - DIRECT
  - INDIRECT: ΔVEGA, SEEGA

  JUPITER SWINGBYS IN COMBINATION WITH BOTH

- ORBIT CAPTURE MODES
  - EARTH-STORABLE RETRO: I_{sp} = 315 sec, f = 0.1332, M_I = 69.9 kg
  - SPACE-STORABLE RETRO: I_{sp} = 370 sec, f = 0.1350, M_I = 154.6 kg
  - AEROCAPTURE: BICONIC AEROSHELL, MODERATE L/D
  - SPIRAL CAPTURE WITH NEP FLIGHT MODE

- PERFORMANCE CALCULATIONS
  - L.V. ADAPTER = 5% M_{INJ}
  - MIDCOURSE NAVIGATION BUDGET: 0.050 km/sec PER PLANET-TO-PLANET LEG
  - ORBIT TRIM MANEUVERS = 0.100 km/sec
  - PROBE/PENETRATOR DEFLECTION MANEUVERS = 0.050 km/sec

Two methods of effecting orbit capture were studied for the missions presented here: (1) impulsive orbit capture, employing chemical retropropulsion for performing the ΔV maneuver, and (2) aerocapture technology. The classical systems assumed are summarized in Table 2. Most recent studies of aerocapture have assumed use of a biconic vehicle with a moderate L/D. This vehicle makes a single deep pass through the atmosphere at the periapsis of its approach trajectory. Enough kinetic energy is removed by aerodynamic drag to capture the vehicle and place it in a transfer orbit to its final operational altitude.

Finally, the specifics of the assumed ΔV budgets, adapter masses, etc. used in making the calculations are summarized as the last item of Table 2.

CANDIDATE MISSIONS

Based on the science objectives outlined previously, 11 potential missions to the five outer planets were identified and examined. These include: four missions to Jupiter and its satellites; two missions to Titan and one mission each to Saturn, Uranus, Neptune and Pluto. The final mission in this set examines the consequences of modifying the Uranus Flyby/Probe spacecraft (a currently identified Core Program mission) into an orbiter/probe vehicle. Characteristics of these candidate missions are summarized in Table 3.

The four missions to Jupiter would: (1) investigate the inner magnetospheric region; (2) investigate atmospheric properties at various locations and depths; (3) establish a monitoring network on the Galilean satellites; and (4) conduct a detailed investigation of Europa. The inner magnetosphere mission would be carried out by a low-altitude (1000 km by 12 R_J) polar orbiter which will allow detailed cross-sectional measurements to be made of the inner magnetosphere and the charged particle radiation belt. The multiprobe mission would deliver four atmospheric probes to various latitudes, longitudes, and depths within Jupiter. Three of these probes would be identical to the Galileo probe with at least one of the three targeted for the Great Red Spot. The remaining probe would be designed to survive to a 1000 bar pressure level and would be targeted at the equator. The third candidate mission would deliver penetrator-type monitoring stations to the three outer Galilean satellites as part of a flyby tour to survey these bodies. The last candidate mission for Jupiter would deliver a Viking-class lander to the surface of Europa while an orbiting vehicle conducts a global survey. This mission could also be carried out at Ganymede or Callisto, but Europa was chosen here because it would be the most challenging mission to complete based on the performance requirements to reach this satellite.

Of the three missions identified for the Saturn system, two would be directed at Titan and one at Saturn itself. The two Titan missions consider alternate means of conducting detailed investigations of the entire satellite (Ref. 21). Each mission would have an orbiter to carry out a planet-wide survey. Both missions would also use entry probes which would deliver surface penetrators and/or buoyant stations with varying levels of sophistication. The buoyant station vehicles would range from simple free-drift balloons to controllable airships. The major difference between these two missions would be in the number and type of entry probes used. The single mission targeted for Saturn itself would use a nuclear electric low-thrust stage to investigate the ring system by essentially hovering above the ring in a non-Keplerian orbit (Ref. 5).

Two missions to Uranus were examined as part of this study. The first of these looked at the currently identified Flyby/Probe mission and estimated the cost to upgrade it into an orbiter/probe mission. While the basic science objectives and instrumentation of the mission would not be changed, this does represent a moderate augmentation due to the increased costs for extended mission operations and the additional propulsion capability or aerocapture vehicle needed for orbit capture. The single new mission to Uranus would deliver three atmospheric probes, designed to survive to 100 bars pressure, into the planet. An orbiter would then continue with a detailed investigation of the planet, its ring system and its satellite system.

Neptune was targeted for a single mission. As part of the scientific investigation to be carried out, a single atmospheric probe would be deployed separately into both Neptune and Triton. Again, an orbiting vehicle (which delivered the probes) would carry out an overall survey of the planet and its satellite system. Two delivery system options were considered for this mission: a NEP option and an aerocapture option. These two options were both carried throughout the study since Neptune is the approximate point at which it becomes more economical to use NEP systems than to use aerocapture for an orbital mission of this type.

A basic assumption made for the single Pluto mission was that it would be the first close-up investigation of this planet and its satellite Charon. A low-thrust NEP stage was chosen since this was the only identified means of delivering a spacecraft to Pluto in a reasonable period of time. This vehicle would deliver a small hard landing probe to the surface of each body and then continue with a detailed survey of this planetary system.

The major characteristics for each of these missions, including trajectory data, injected mass requirements and estimated cost, have been summarized in Table 3.

SUMMARY

The wide range of spacecraft and missions just discussed indicates the breadth of knowledge still to be obtained from exploration of the outer planets. The Committee on Planetary and Lunar Exploration (COMPLEX) of the Space Science Board has established a number of scientific objectives to help fill some of the gaps in the present understanding of these planets. The Saturn Orbiter/Titan Probe mission (Cassini) and the Saturn Flyby/Saturn Probe mission, which have both been included in the SSEC Core Program, will meet several of these objectives. The purpose of this study has been to examine possible mission concepts to be carried out in the 1995-2015 time period which will fulfill the remainder of the objectives. Eleven candidate missions were identified and examined. It was found that existing hardware elements and concepts could be used to carry out most of these missions. There were

Table 3

SUMMARY OF CANDIDATE MISSION CHARACTERISTICS

MISSION	FLIGHT MODE	TRAJECTORY TYPE	LAUNCH VEHICLE	CAPTURE MODE	FLT TIME (YRS)	INJ. MASS (kg)	TOTAL COST ($M)
1. JUPITER INNER MAGNETOSPHERE/POLAR ORBITER	BALLISTIC	DIRECT	CENTAUR(G')	E-S RETRO	2.2	2030	609
2. JUPITER DEEP PROBE/MULTIPROBE	BALLISTIC	DIRECT	OOA CENT(G')/ CENT(G)	E-S RETRO	1.9	4450	1401
3. GALILEAN SATELLITE PENETRATOR NETWORK	BALLISTIC	DIRECT	OOA CENT(G')/ CENT(G)	E-S RETRO	3.5	3866	1174
4. EUROPA ORBITER/LANDER	BALLISTIC	DIRECT	OTV/OTV	E-S RETRO	3.5	6120	1466
5. TITAN ORBITER/PENETRATOR NETWORK	BALLISTIC	DIRECT	OOA CENT(G')/ STAR 48	AEROCAPTURE	4.5	1960	1247
6A. TITAN ORBITER/BUOYANT STATION (ORBITER)	BALLISTIC	DIRECT	OOA CENT(G')/ STAR 48	AEROCAPTURE	4.5	1885 ORB	2481
6B. TITAN ORBITER/BUOYANT STATION (PROBE CARRIER)	BALLISTIC	DIRECT	OOA CENT(G')/ CENT(G)	FLYBY	4.2	2730	
7. SATURN RING ROVER	NEP	DIRECT	SHUTTLE	SPIRAL	10.4	16736	1245
8. URANUS ORBITER/PROBE	BALLISTIC	J/U	OOA CENT(G')/ STAR 48	E-S RETRO	11.0	1946	933
8A. URANUS ORBITER/PROBE	BALLISTIC	J/U	CENTAUR(G')	AEROCAPTURE	9.0	5163	985
9. URANUS ORBITER/MULTIPROBE	BALLISTIC	J/U	OTV/OTV	AEROCAPTURE	6.0	4955	1403
10. NEPTUNE ORBITER/DUAL PROBE	BALLISTIC	J/N	OTV/OTV	AEROCAPTURE	7.0	3947	1562
10A. NEPTUNE ORBITER/DUAL PROBE	NEP	DIRECT	OTV/OTV	SPIRAL	8.3	17649	1799
11. PLUTO ORBITER/LANDER AND CHARON LANDER	NEP	DIRECT	OTV/OTV	SPIRAL	7.7	14648	1489

several exceptions which occurred due to the nature of the mission involved. Some of these required unique solutions such as the airship concept used at Titan. Others required hardware, such as NEP or aerocapture, to fulfill mission needs which would have a wider range of application in the future. The cost for these new missions was found to range from $609M to $2481M in FY 1986 dollars. All of the mission concepts tend to be ambitious in scope and, in most cases, could be split into separate missions to reduce individual mission costs. There were no identified technological impediments to meeting any of the desired science objectives.

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ACRONYMS AND ABBREVIATIONS

AACS	attitude and articulation control system
A-C	aerocapture
BOL	beginning of life
CCD	charged couple device
CDH	command and data handling system
COMPLEX	Committee on Planetary and Lunar Exploration
DARPA	Defense Advanced Research Project Agency
DOE	Department of Energy
DSS	data storage system
ΔVEGA	ΔV Earth gravity-assist
E-S	Earth-storable
ESA	European Space Agency
GPHS	general purpose heat source
L/D	lift-to-drag ratio
MHW	multi-hundred watt
NEP	nuclear electric propulsion
OMS	orbital maneuvering system
OOA	on-orbit assembly
OOF	on-orbit fueling
OTV	Orbital Transfer Vehicle
RCS	reaction control system
RFS	radio frequency system
RTG	radioisotope thermal generator
SEEGA	solar electric Earth gravity-assist
SEP	solar electric propulsion
S-S	space-storable
SSEC	Solar System Exploration Committee

1. INTRODUCTION

1.1 Background

The outer planets, from Jupiter with its diverse assemblage of 16 satellites to the virtual double planet of Pluto and Charon, exhibit a wide range of natural phenomena which differ greatly from anything found on Earth or among the inner planets. This variety has kept the science community active over the past several centuries attempting to determine the nature of these bodies, comparing them to one another to better understand their differences and similarities. The comparatively recent introduction of interplanetary spacecraft to this investigative process has done much to reveal the nature of these planets but has also generated an entirely new set of queries as each world is examined in more detail.

The exploration of a planet, for the purpose of resolving the many questions concerning it, can be divided into three major phases. The first of these phases is the initial reconnaissance of the planet to obtain a general accounting of what is to be found there and establish the type of questions to be examined by subsequent vehicles. The second phase involves a global characterization of the planet which results in a more specific accounting of the planet's attributes (the exploration phase). Finally, a phase of intensive study begins during which each new vehicle focuses on a single attribute or set of related attributes in order to gain the best possible understanding of the planet.

Interplanetary missions to the outer planets have been relatively few in number, compared to the spacecraft orbiting the Earth or exploring the inner solar system. The first vehicles to cross the asteroid belt into the region of the outer planets were Pioneers 10 and 11. The major objective of these missions was to conduct the initial reconnaissance of the planet Jupiter and its four Galilean satellites. A favorable Jupiter gravity assist allowed Pioneer 11 to be retargeted for Saturn, where it carried out a similar mission. Voyagers 1 and 2 used the information returned from Pioneers 10 and 11 to continue the reconnaissance and begin the global characterization of both Jupiter and Saturn. Again a favorable geometry has allowed Voyager 2 to be retargeted for a flyby encounter with both Uranus and Neptune. The last of this generation of spacecraft will be Galileo which is to continue the global characterization of the Jupiter system, including the atmosphere (via an entry probe and remote sensing instruments) and the major satellites. Figure 1-1 summarizes the accomplishments of these five vehicles. Even with this level of accomplishment, many questions still remain outstanding and new issues are arising from Earth-based discoveries including the rings at Uranus, Pluto's moon Charon, and evidence for an atmosphere on the Neptunian satellite Triton. However, there has been a lack of new missions to follow up on these questions and discoveries. The Pioneers and Voyagers were all launched in the 1970's and Galileo is scheduled to be launched in 1986 with no other funded missions to follow it.

This lack of activity has prompted NASA to seek a method of sustaining planetary exploration over the long term. The Solar System Exploration Committee of the NASA Advisory Council was formed in 1980 to devise a sustainable, fiscally realizable long-range program of planetary missions (Ref. 1). Specifically, the Committee was charged with the task of translating specific scientific recommendations for solar system exploration from the Space Science Board of the National Academy of Sciences (Ref. 2) into a realistic set of exploration missions. This set of missions would extend through the end of the century and would occur with a frequency which would allow the economies of heritage in software and hardware to be fully utilized.

Findings from this Committee's efforts (Ref. 1) resulted in a recommendation for sustained annual funding in three areas: (1) research and analysis; (2) mission operations and data analysis; and (3) a set of high priority solar system exploration missions collectively known as the Core Program. Missions to the inner planets, outer planets and primitive bodies are all represented within this program. Four missions to the outer planets have been identified for implementation as part of the program, including:

- Titan Probe/Radar Mapper
- Saturn Orbiter
- Saturn Flyby/Saturn Probe
- Uranus Flyby/Uranus Probe

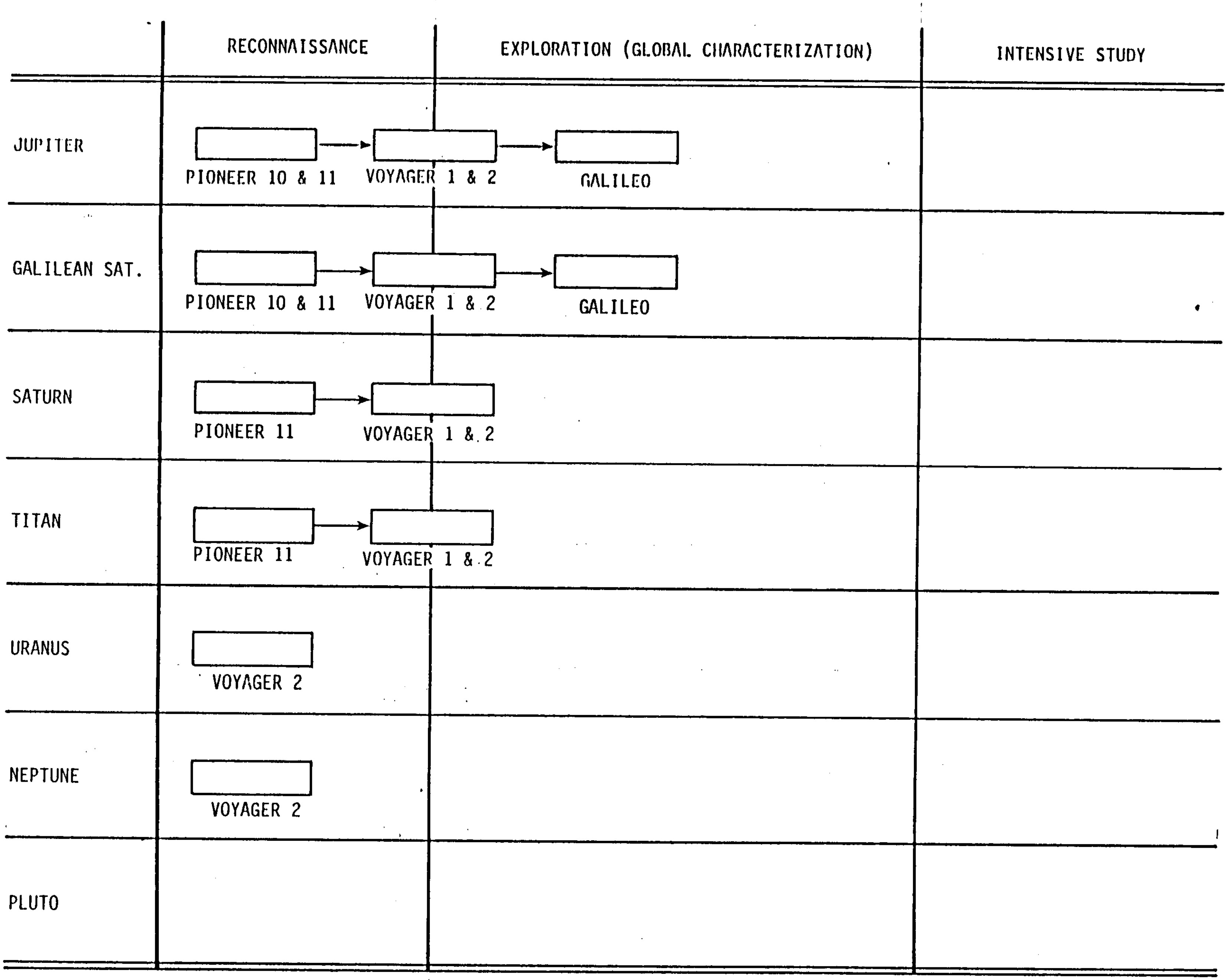

Figure 1-1. Current Status of Outer Planet Investigation by All Previous U.S. Missions

In the time since the Core Program was originally formulated, the possibility of combining two of these missions into an international effort has been discussed. Under this option the Titan Probe and Saturn Orbiter would be merged into a single mission, known as Cassini, as a joint undertaking by NASA and the European Space Agency (ESA).

Two other outer planet missions were also identified as highly desirable. However, due to the flight times associated with these missions and the higher priority assigned to other missions, it was not possible to incorporate them into the Core Program before the year 2000 limit placed on the SSEC. These missions include:

- Neptune Flyby/Probe
- Pluto Reconnaissance Flyby.

Recalling the information presented in Figure 1-1, Figure 1-2 illustrates how these six high priority missions will expand knowledge of the outer planets.

The inability of the Core Program to accommodate the Neptune and Pluto missions points out the fact that this Program is only a beginning toward meeting many of the objectives raised by the Space Science Board. Indeed, the SSEC recognized that there would be missions of larger scope or greater technical difficulty than those in the Core Program which would be required in order to satisfy some of the recommendations of the Space Science Board. A sample return mission from any body beyond the Earth-Moon system is but one example of this type of mission. The SSEC categorized these missions as augmentations to the Core Program which would be pursued as funding permitted.

1.2 Study Objectives

Using the background described above as a framework, the purpose of this study is to develop a "broad brush" picture of advanced mission concepts for outer planet exploration in the 1995-2015 time frame. The set of missions developed over the course of this study could satisfy most, if not all, of the Space Science Board recommendations not covered by the proposed Core Program. Each mission would have a sufficiently detailed description of science

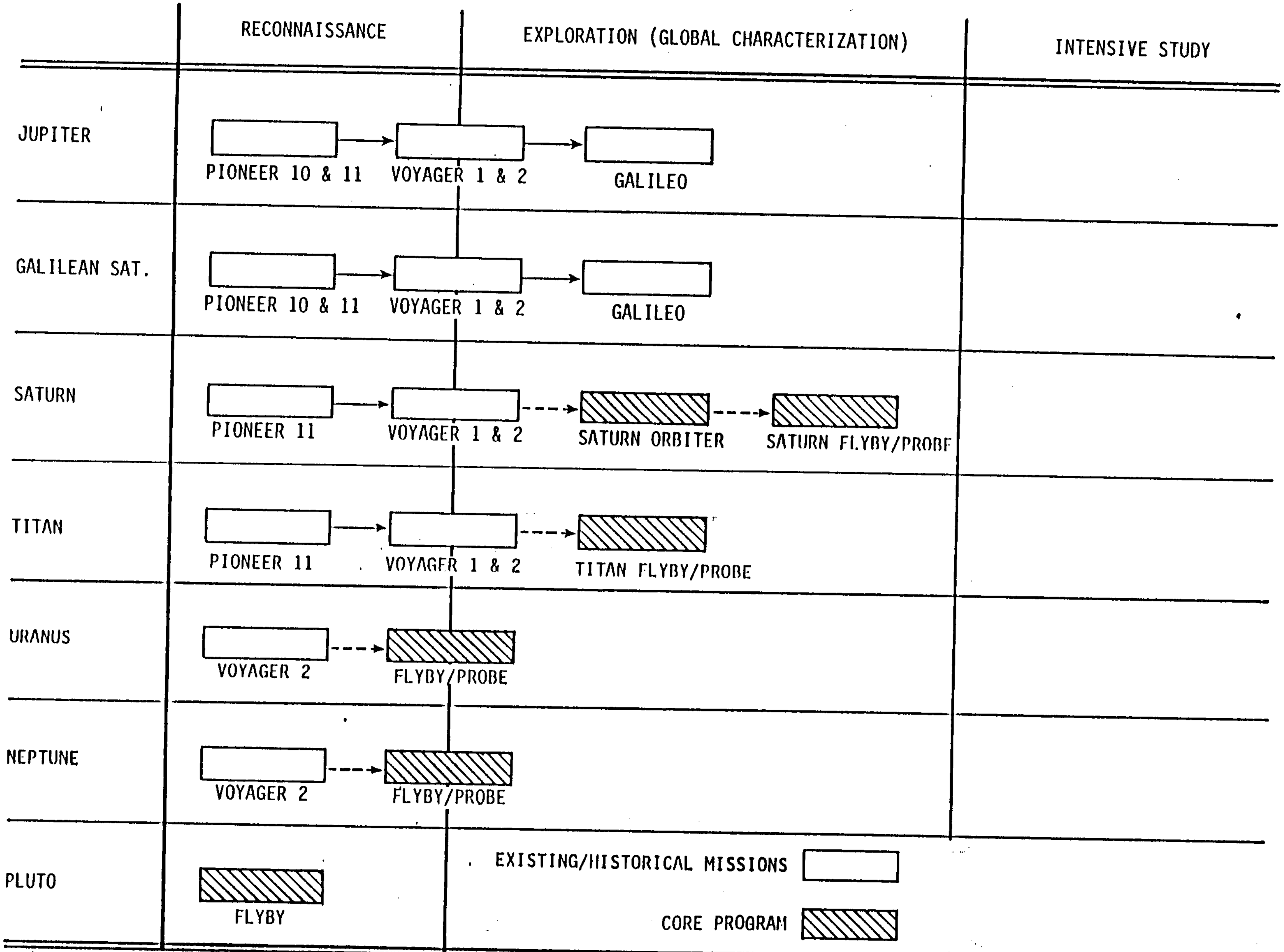

Figure 1-2. Status of Outer Planet Investigation through the Completion of All Identified SSEC Core Program Missions

objectives to allow candidate science instrument payloads, spacecraft functional requirements and mass breakdowns to be defined. This information will allow performance/flight time trades to be conducted and key operational characteristics or design issues to be identified. A cost estimate for each mission may then be made, reflecting both the heritage from Core Program missions and new technology required to meet the science objectives.

To accomplish this objective, this study was divided into four major tasks, which are:

1. Define the science objectives and associated missions (with appropriate science payloads).
2. Identify candidate hardware to support each mission.
3. Conduct performance trade studies.
4. Summarize all elements of each mission and perform an appropriate cost estimate.

The first task involves the review of the recommendations for exploration of the outer planets from the Committee on Planetary and Lunar Exploration (COMPLEX) of the Space Science Board (Ref. 2). These recommendations are compared to the objectives of the Core Program missions to help minimize any overlap between the Core Program and missions to be studied here. This results in a set of candidate missions with appropriate science payloads.

The second task uses the science objectives and candidate payloads as a basis to identify the types of hardware needed to support these missions. The technology readiness or heritage from previous missions can then be assessed and the appropriate combination of support hardware can be assigned to each mission. The final phase of this task is to devise a mass statement listing the science payload and support hardware for each mission.

The third task examines trajectory performance issues during the 1995-2015 time frame with emphasis on the best opportunity year for each mission. One or two trajectories exhibiting the best performance for the mission under consideration are used to select an appropriate upper stage vehicle.

The final task is to gather together all pertinent information for each mission from the three previous tasks. A cost estimate for each mission is then made based on the hardware elements and trajectory characteristics needed to complete the mission. This information is summarized and presented in a format common to the entire mission set.

2. SCIENCE OBJECTIVES

2.1 The Core Program

The portion of the SSEC Core Program which addresses missions to the outer planets was established in order to carry out a systematic investigation of the many ring systems, satellite systems and planetary atmospheres of these gas giants. This program will thus support a rigorous level of investigation and continuity of exploration of the outer solar system for the 1990-2000 time frame. It will also allow major advances to be made in answering many basic questions about these planetary bodies. The selected candidate missions of the Core Program are:

1. **Saturn Orbiter/Titan Probe:** Now known as Cassini (a joint NASA/ESA effort), this mission will characterize Titan's atmospheric composition, structure and environment while providing a Galileo-type tour of Saturn's ring system, satellites, magnetosphere and atmospheric dynamics.

2. **Saturn Flyby/Probe:** This mission will characterize Saturn's atmosphere with an entry probe and provide flyby science of the Saturn system.

3. **Uranus Flyby/Probe:** This mission will provide a characterization of Uranus' atmosphere as well as conducting flyby science of the Uranus system.

4. **Neptune Flyby/Probe:** This mission is identical in scope to the Saturn and Uranus Flyby missions but would not be implemented until the 2000-2005 time frame.

5. **Pluto Flyby:** This mission would characterize the Pluto system but would not be launched before the 2000-2005 time frame.

These missions are designed to use the Mariner Mark II spacecraft and utilize, as much as possible, existing hardware from previously flown missions.

2.2 The Augmented Program

The Augmented Program will upgrade the level of solar system exploration of selected targets from the exploration phase to the intensive study phase. Each mission of the Augmented Program (referred to in this report as "Augmentation missions") will be more technologically challenging than those of the Core Program. These missions will focus on each of the outer planets by utilizing orbiters, probes, balloon stations, penetrators and landers. The following is a list of candidate Augmentation missions that achieve the goals of intensive study in the 1995-2015 time frame:

1. Jupiter Inner Magnetosphere/Polar Orbiter;
2. Jupiter Deep Probe/Multiprobe;
3. Galilean Satellite Penetrator Network;
4. Europa Orbiter/Lander;
5. Titan Orbiter/Penetrator Network;
6. Titan Orbiter/Buoyant Station;
7. Saturn Ring Rover;
8. 1995 Uranus Orbiter/Probe;
9. Uranus Orbiter/Multiprobe;
10. Neptune Orbiter/Dual Probe; and
11. Pluto Orbiter/Lander and Charon Lander.

Each of these missions will be described in detail separately in the following section.

2.3 Candidate Science Mission Summaries

2.3.1 Introduction

Each mission is presented in a one or two page format outlining the science rationale and scientific objectives for the mission, a list of science questions to be addressed, instrumentation to be included in the mission and expected results of that instrumentation, and a brief mission scenario describing what will be done when the spacecraft reaches its target.

2.3.2 Jupiter Inner Magnetosphere/Polar Orbiter

Science Rationale/Objectives. A Polar Orbiter at Jupiter would allow detailed cross-sectional measurements to be made of the intense inner magnetosphere and radiation belts. In addition, this vehicle would observe Io's role in the development of auroral activity in Jupiter's atmosphere and the Io plasma torus. Specific objectives include:

- Determine the density, composition and energy of magnetosphere particles;
- Determine the large-scale structure and rotation of the magnetosphere;
- Determine time-dependent phenomena and relation of magnetosphere to Io, other satellites, orbiting gases and plasmas;
- Determine the nature of auroral activity; and
- Determine the nature of electromagnetic emission.

Compelling Science Questions

1. Does Io affect (or control) Jupiter's auroral activity?
2. How does the magnetospheric structure vary with time?
3. What is the nature of Jupiter's electromagnetic emission?
4. How does the solar wind interact with the magnetosphere at the poles?
5. Is the size of Jupiter's ring system a result of the harsh environment of the inner magnetosphere?

Instruments and Expected Results

MAGNETOMETER: measure the structure of the inner magnetosphere, its rotation and interaction with Io

PLASMA DETECTOR: measure low energy particles and hot ionized gas trapped in the inner magnetosphere and Io's plasma torus

PLASMA WAVE INSTRUMENT: measure waves generated inside the magnetosphere and determine electromagnetic emission

ENERGETIC PARTICLE DETECTOR: measure the composition, distribution and energy spectra of high energy particles trapped in the magnetosphere

COSMIC RAY DETECTOR: measure release of cosmic rays by trapped particles of the inner magnetosphere; study composition of radiation belt

DUST DETECTOR: measure mass, velocity and charge of particles inside the magnetosphere

ULTRAVIOLET SPECTROMETER: measure light-scattering properties of the upper atmosphere of Jupiter

ION MASS SPECTROMETER: measure distribution and concentration of positively charged ions in the upper atmosphere

NEUTRAL MASS SPECTROMETER: measure densities of neutral atoms and molecules in the ionosphere

ELECTRON TEMPERATURE PROBE: measure thermal properties of the ionosphere

IMAGING PHOTOPOLARIMETER: measure the vertical distribution of cloud and haze particles; observe auroral activity; observe Io's sodium cloud and plasma torus; observe Jupiter's ring system.

Mission Scenario. The Cruise phase of the Jupiter Inner Magnetosphere/Polar Orbiter mission will begin approximately 30 days after launch and continue until 30 days prior to orbit insertion. During this phase, cruise science activities will conduct particles and fields measurements of the interplanetary medium for 20 hours once every two months.

The Far Encounter phase will begin 30 days prior to orbit insertion. At this time, the particles and fields instruments will be operating at a high data rate capacity, taking measurements once every hour. From this point until orbit insertion, the spacecraft instruments will monitor the location of the bow shock and measure the environment between the inner and outer magnetospheric boundaries (a distance of approximately 6 million km).

The spacecraft will be inserted into a polar orbit about Jupiter so that it will pass through the inner magnetosphere region at a minimum distance of 1.016 R_J. Once in orbit, the instruments will operate continuously at a high data rate capacity, mapping the dynamics, structure, and environment of the inner magnetosphere.

2.3.3 Jupiter Deep Probe/Multiprobe

Science Rationale and Objectives. The atmosphere of Jupiter is the most turbulent and complex of any planet in the solar system. Although the Galileo

Probe will provide the first direct examination of the Jovian atmosphere, only one small area will be examined on a world more than 10 times the size of the Earth. Clearly, in order to understand the meteorology, composition and driving forces of this atmospheric dynamo a more global investigation must be conducted. The Jupiter Deep Probe/Multiprobe is a mission which will address these problems by examining the atmosphere at different latitudes and longitudes and probing at much greater depths than the Galileo mission.

The science objectives include:

- **Multiprobe** (Long duration): Characterize the dynamics, structure and composition of the Jovian atmosphere at three widely separate locations down to a level of 20 bars. All three locations will be in the southern hemisphere with special emphasis on the Great Red Spot.
- **Deep Probe**: Characterize the dynamics, composition and structure of Jupiter's atmosphere at the equatorial zone down to a level of 1000 bars.

Compelling Science Questions

1. Do complex molecules exist deep within Jupiter's atmosphere?
2. What is the source of the colors of the clouds?
3. What is the nature and structure of the Great Red Spot?
4. What is the convective structure of the atmosphere?
5. What meteorological conditions prevail?
6. What is the driving force behind the Great Red Spot and other large cyclonic disturbances?

Instruments and Expected Results

Multiprobe (Long Duration)

ATMOSPHERIC STRUCTURE INSTRUMENT: measure temperature, density, pressure and molecular weight of the atmosphere

HELIUM ABUNDANCE DETECTOR: measure the ratio of H/He in the atmosphere

CLOUD PARTICLE SIZE SPECTROMETER: measure cloud particle size, shape and density thus providing a vertical profile of particle concentration

NET FLUX RADIOMETER: measure the radiative energy from Jupiter inside the atmosphere along with incoming solar energy

NEUTRAL MASS SPECTROMETER: measure the chemical composition of the atmosphere

LIGHTNING AND RADIO EMISSIONS DETECTOR: determine the presence of lightning through generated radio emissions

Deep Probe

ATMOSPHERIC STRUCTURE INSTRUMENT: measure temperature, density, pressure and molecular weight of the atmosphere

HELIUM ABUNDANCE DETECTOR: measure the H/He abundance ratio

NEPHELOMETER: measure the size of cloud particles and the location of layers in the atmosphere

NEUTRAL MASS SPECTROMETER: measure the chemical composition of the atmosphere

GAS CHROMATOGRAPH: search for the presence of organic molecules and identify noble gases present

IR RADIOMETER: measure thermal flux as a function of altitude; detect cloud layers and water vapor

Mission Scenario. The deep probe will be released from the probe bus 150 days prior to arrival at Jupiter. This will allow the probe to begin transmitting data just as the orbiter is arriving at Jupiter and before the orbit insertion maneuver. After entry, the probe's accelerometers (part of the atmospheric structure instrument) will begin monitoring and measuring the deceleration forces acting upon the probe beginning at an altitude of 450 km. A drogue chute will pull the main chute out for deployment at an altitude of 45-50 km. At this point in the mission, all the science instruments are functioning. At the 10 bar level, the deep probe will separate from a relay probe still attached to the main parachute. This will facilitate data transmission between the probe and the bus. After separation, the probe will free-fall, sampling the environment as it falls, down to a level of 1000 bars.

The orbiter/bus will then conduct an orbit insertion maneuver to place it in an elliptical orbit. Each remaining probe will be independently launched and targeted to a separate part of the planet.

The candidate targets include:

1. The Great Red Spot - 20°S latitude;
2. Across the day/night terminator - also at 20°S latitude; and
3. 60°S latitude.

As each probe is released, the lightning and radio emissions detector will begin a pre-entry search for lightning-generated emissions. This data will be stored on-board and retransmitted after entry is complete.

The accelerometers (part of the atmospheric structure instrument) will begin measuring the deceleration forces generated by the probe's plunge through the atmosphere, at an altitude of 450 km above the cloud tops. At approximately the 80 km level, a drogue chute will deploy, pulling out the main parachute, releasing the probe from its heat shield and thus slowing its rate of descent. By 45 km altitude, all the science instruments will be fully operational conducting measurements down to the 20 bar level. This phase should last about one hour.

The multiprobe aspect of the mission can be enhanced using a lifting structure (i.e., kite, airfoil, etc.) instead of the parachute that would allow the probe to remain suspended at one level for an extended period of time.

2.3.4 Galilean Satellite Penetrator Network

Science Rationale/Objectives. A seismic network deployed by penetrators on one or all of the Galilean satellites would allow detailed investigation of their internal structure and help to determine the degree of volcanic and tectonic activity for each of these planet-sized worlds. Specific objectives are:

- Monitor the seismic activity of target and satellite to develop a model of the internal structure;
- Determine subsurface chemical and mineralogical composition;
- Measure local magnetic field and interaction of satellite surface with the magnetosphere;

- Determine heat flow of the satellite; and
- Measure physical properties of surface material.

Compelling Science Questions

1. What are the internal structures of the Galilean satellites?
2. How do the Galilean satellites interact with Jupiter's magnetosphere?
3. What is the subsurface geochemistry?
4. Do the Galilean satellites possess any intrinsic magnetic field?
5. How thick is the satellite crust?

Instruments and Expected Results

3-AXIS SEISMOMETER: monitor and measure intensity of seismic activity in order to establish a model of internal structure, composition and homogeneity

ALPHA-PROTON BACKSCATTER/X-RAY FLUORESCENCE SPECTROMETER: measure subsurface elemental composition and abundance

TEMPERATURE SENSORS: conduct heat flow measurements from below the surface

IMPACT ACCELEROMETER: determine subsurface stratigraphy by measuring rate of deceleration and depth of penetration

MAGNETOMETER: measure local magnetic field intensity; monitor interaction between satellite surface and Jovian magnetosphere

FACSIMILE IMAGER: provide imaging of landing area for site and regolith characterization

WATER DETECTOR AND HYDRATED MINERAL ANALYZER: measure free and chemically bound water content below the surface

Mission Scenario. The spacecraft is inserted into Jupiter orbit utilizing a Ganymede swingby. There are several mission options which may be performed while in orbit. At one extreme, each satellite is targeted for one penetrator. The spacecraft establishes resonance with the target satellite and releases the penetrator. The bus remains in the satellite's orbital plane, retrieving data on each satellite pass. Once all the desired data have been gathered, the spacecraft is pumped up or down to the next target satellite and the scenario is repeated. At the other extreme, one satellite is targeted for all the penetrators with the bus entering orbit about the target satellite (in this case Europa, Ganymede or Callisto), collecting data continuously.

2.3.5 Europa Lander/Orbiter

Science Rationale/Objectives. The surface of Europa is one of the smoothest surfaces in the solar system. This may imply possible volcanic activity or upwelling through surface cracks. Specific objectives for this mission include:

- **Lander Objectives**
  - o Investigate the internal properties of Europa by measuring degree of seismic activity;
  - o Provide surface imagery of the landing site;
  - o Observe Jupiter at favorable phase angles;
  - o Observe the interaction between Europa's surface and Jupiter's magnetosphere, including any potential sheathing effects by Europa's own indigenous magnetic field (if any);
  - o Measure local magnetic field strengths;
  - o Analyze elemental composition of the surface; and
  - o Determine the mineralogy and petrology of surface material.

- **Orbiter Objectives**
  - o Map elemental and mineralogical distribution on the satellite surface;
  - o Conduct extensive surface geology observations;
  - o Measure mass and radius of the satellite;
  - o Monitor interaction of the satellite with the Jovian magnetosphere; and
  - o Determine thermophysical properties of Europa (i.e. heat flow, temperature conductivity, temperature, porosity, etc.).

Compelling Science Questions

1. Why does Europa have the smoothest surface in the solar system?
2. Is Europa volcanically or tectonically active?
3. What is the nature and composition of its surface?

4. What is the nature of the many streaks on Europa's surface?

Instruments and Expected Results

Lander

MULTISPECTRAL IMAGING: perform imaging of the landing site, observation of Jupiter, experiment documentation and support

3-AXIS SEISMOMETER: monitor and measure intensity of seismic activity at landing site

MAGNETOMETER: measure local magnetic field strength; measure interaction between satellite's surface and Jovian magnetosphere

ALPHA-PROTON BACKSCATTER/X-RAY FLUORESCENCE SPECTROMETER: measure elemental abundance and composition of sampled surface material

SCANNING ELECTRON MICROSCOPE/MICROPROBE: conduct analysis of mineralogical and elemental composition (trace and minor elements) of sampled surface material

PETROGRAPHIC MICROSCOPE: provide mineral identification of selected sample material

TEMPERATURE SENSOR: measure heat flow values at the surface

SAMPLER ASSEMBLY: obtain surface samples for analysis; test physical properties of surface material

Orbiter

MULTISPECTRAL IMAGING (CCD): conduct high resolution photography of Europa's surface; landing site selection; observe activity at Io

MAGNETOMETER: measure satellite's magnetic field; measure interaction between satellite and Jovian magnetosphere

MULTICHANNEL MICROWAVE RADIOMETER: map thermophysical properties of the surface including heat flow and surface temperature

X-RAY SPECTROMETER: map elemental composition of the surface (heavy shielding required)

NEAR-IR/VISIBLE MAPPING REFLECTANCE SPECTROMETER: map mineralogy of satellite surface; map areas of particular interest based on Galileo observations

ENERGETIC PARTICLE DETECTOR: measure high energy electrons, protons and heavy ions within Jupiter's magnetosphere

RADAR ALTIMETER/DOPPLER TRACKING: measure intensity of gravity field, surface roughness and other geophysical characteristics

Mission Scenario. The spacecraft enters Jupiter orbit using a Ganymede swingby. The orbit is pumped down to Europa's radius about Jupiter using multiple flybys of other Galilean satellites. The spacecraft is then inserted into orbit about Europa. The imaging system and mapping spectrometer are used to search for the most interesting landing site. When an appropriate site has been selected, the lander is deployed. Upon separation, the orbiter will begin conducting full scale global mapping observations and experiments. Once the lander is operational, the seismometer and magnetometer will begin continuous operation while the geochemical experiments and heat flow sensor will operate periodically for the duration of the mission.

2.3.6 Titan Orbiter

Science Rationale/Objectives. An orbiter at Titan (Ref. 21) will allow long-term observation and analysis of its atmosphere as well as provide clues to the nature of its surface. Specific objectives include:

- Conduct radar mapping of Titan's surface to provide images and topography;
- Conduct spectroscopic observations of the atmosphere and thermal radiation by the surface;
- Conduct radio occultation experiments of the upper atmosphere to provide temperature and density profiles;
- Obtain images of the surface in the near IR;
- Determine physical characteristics of Titan through gravity perturbation;
- Determine Titan's magnetic field strength (if any) and study its interaction with Saturn's magnetosphere;
- Perform low orbit aeronomy experiments; and
- Observe the amount of material entering the atmosphere.

Compelling Science Questions

1. What is the nature of Titan's surface?

2. Is there volcanic activity on Titan?
3. How does Titan interact with Saturn's magnetosphere as it traverses this field during its orbit?
4. Why is the southern hemisphere atmosphere brighter than the northern hemisphere?
5. Does Titan's atmosphere undergo seasonal changes?
6. Is Titan's surface chemistry responsible for its atmosphere?

Instruments and Expected Results

MAGNETOMETER: measure Titan's magnetic field and observe its interaction with Saturn's magnetosphere

PLASMA WAVE INSTRUMENT: investigate energy waves generated inside Saturn's magnetosphere in the vicinity of Titan

PLASMA DETECTOR: conduct low orbit aeronomy of Titan's upper atmosphere; measure low energy particles in the vicinity of Titan inside Saturn's magnetosphere

NEUTRAL MASS SPECTROMETER: conduct low orbit aeronomy measurements

DUST PARTICLE ANALYZER: determine the amount of material entering the atmosphere; determine the composition and density of this particulate matter

UV SPECTROMETER: measure the amount of UV light scattered by Titan's atmosphere

THERMAL IR SPECTROMETER: measure and map the thermal radiation from Titan's surface

NEAR-IR IMAGER: obtain images of Titan's surface in the near-IR spectral range

RADAR MAPPER: map the surface of Titan with the same resolutions as the Venus Radar Mapper spacecraft

RADIO OCCULTATION, TRACKING: provide cross-sectional temperature and density profiles of the atmosphere and measure effects of Titan's gravity field on the spacecraft's orbit

Mission Scenario. The spacecraft will either enter polar orbit around Titan after it has entered orbit around Saturn or it will use the aerocapture system for direct orbit insertion about Titan. Once in orbit, the spacecraft would begin global mapping of the atmosphere and surface in addition to serving as a data link for any landers, balloons or penetrators at the surface.

2.3.7 Titan Penetrator Network

Science Rationale/Objectives. A penetrator system would be useful in monitoring the long-term variations in Titan's weather and meteorology as well as in establishing a model of Titan's interior through a seismic network. Specific objectives include:

- Measure physical properties of the surface upon impact;
- Monitor seismic activity in order to develop a model of Titan's internal structure;
- Conduct chemical analysis of subsurface material;
- Determine subsurface stratigraphy;
- Conduct *in situ* weather observations over a long period of time; and
- Measure local magnetic field strengths.

Compelling Science Questions

1. What is Titan's internal structure?
2. Does Titan have a "bedrock" equivalent below its surface?
3. Do silicates comprise any portion of Titan's geochemistry?
4. If ice exists on Titan, is there gas trapped inside?

Instruments and Expected Results

3-AXIS SEISMOMETER: monitor seismic activity and record intensities

HEAT FLOW SENSORS: measure heat flow from the satellite's surface

ALPHA-PROTON BACKSCATTER/X-RAY FLUORESCENCE SPECTROMETER: conduct elemental analysis of subsurface material

METEOROLOGY EXPERIMENT: measure temperature, pressure and wind velocity of the atmosphere

IMPACT ACCELEROMETER: measure deceleration through Titan's surface (information will be used to develop model of stratigraphy)

FACSIMILE IMAGER: ·characterize geomorphology of landing site

MAGNETOMETER: measure and monitor local magnetic field intensities and variations

WATER DETECTOR: search for the presence of water in subsurface material

Mission Scenario. Upon entering orbit, three penetrators will be deployed to land at three widely separated areas on the surface. This will allow for a global characterization of weather conditions and geochemistry as well as providing a base for a large seismic array to model the interior.

2.3.8 Titan Buoyant Station

Science Rationale/Objectives. A buoyant station at Titan (Ref. 4) will provide a long term in situ floating laboratory to conduct a detailed analysis of the composition, structure and dynamics of Titan's complex atmosphere. Specific objectives are:

- Observe atmospheric circulation and weather conditions on a planet-wide basis;
- Image surface and map compositional differences of surface;
- Conduct on-board chemical and elemental analysis of surface samples;
- Deploy sonar buoys in lakes or seas; and
- Determine compositional differences between northern and southern hemispheres, if any.

Compelling Science Questions

1. What is the gas content and concentration of Titan's atmosphere?
2. Is the surface of Titan responsible for its dense atmosphere?
3. Are there lakes and seas on Titan's surface?
4. What is the elemental content of the surface?
5. Are there winds in Titan's atmosphere?
6. What constituent in the atmosphere causes the northern hemisphere to be darker than the southern?
7. Is lightning present in the atmosphere?
8. What is the meteorology of Titan?

Instruments and Expected Results

CLOUD PARTICLE SIZE SPECTROMETER: measure cloud particle size, shape and density, thus providing a vertical profile of particle concentration

GAS CHROMATOGRAPH/MASS SPECTROMETER: determine the composition of the lower atmosphere and search for organic compounds

ATMOSPHERIC STRUCTURE INSTRUMENT: measure the temperature, density, pressure and molecular weight of the atmosphere

LIGHTNING DETECTOR: determine the presence of lightning through generated radio emissions

NET FLUX RADIOMETER: measure the radiative energy of the atmosphere along with incoming solar energy

AEROSOL SAMPLE COLLECTOR: analyze the falling aerosol particles for composition

RADAR ALTIMETER: monitor the height of the buoyant station; develop a map of local surface roughness

NEAR-IR IMAGER: image the surface in the near IR to characterize surface topography and conditions

IR SPECTROMETER: map elemental composition of the surface

SURFACE SAMPLER ASSEMBLY: collect samples from the surface for analysis on-board the buoyant station

Mission Scenario. After the probe enters Titan's atmosphere, a parachute is deployed at 100 km altitude immediately after which the balloon is deployed and gas filling operations commence. By the time the balloon reaches an altitude of 5 km, it is completely filled and full scale observations may begin. While floating, the buoyant station will periodically sample the surface for on-board analysis. The buoyant station may also have the capability of deploying sonar buoys and weather stations.

2.3.9 Saturn Ring-Rover

Science Rationale/Objectives. In order to develop any models regarding the origin and evolution of the Saturn system, a detailed investigation of its ring system is necessary. Specific objectives of this mission (Ref. 5) are:

- Determine the size, shape, and spatial distribution of the ring particles as well as their dynamic, electric and optical properties;
- Determine the ring thickness;
- Determine the chemical and elemental composition of the ring particles and assess any variations as a function of distance from Saturn;
- Observe the ring's interaction with Saturn's magnetosphere and gravity field as well as other effects caused by thermal forcing functions and meteoroid activity; and
- Observe interaction of outer rings with "shepherding" satellites.

Compelling Science Questions

1. How were Saturn's rings formed?
2. What is the composition of the rings?
3. Are the rings still evolving ?
4. How do the rings interact with their environment?
5. What is the nature of the ring particles?
6. Why are there so many ringlets instead of one broad ring?
7. What is the nature of the ring spokes?
8. Why are broad rings unique only to Saturn and not the other gas planets?

Instruments and Expected Results

MULTISPECTRAL IMAGER: image the ring system; determine particle size, shape, distribution and optical properties

MAGNETOMETER: measure interaction of the rings with the magnetosphere; map Saturnian magnetosphere

PLASMA DETECTOR: measure low energy particles and ionized gas within the ring system

DUST ANALYZER: determine the size, speed, composition and charge of small particles within the ring system

SCANNING ELECTRON MICROSCOPE/PARTICLE ANALYZER: image individual dust grains of < 40 nm and perform x-ray elemental analysis at a resolution of 2 μm

NEAR-IR VISIBLE MAPPING REFLECTANCE SPECTROMETER: determine and map the mineralogy of particles including ices, clathrates and condensates

MULTICHANNEL MICROWAVE RADIOMETER: determine the bulk absorption coefficient of ring material

ION MASS SPECTROMETER: determine the neutral gas, ion and dust composition of particles

RADAR: aid in navigation within the ring system; determine ring thickness; measure the physical roughness of larger particles

ENERGETIC PARTICLE DETECTOR: detect the high energy particles in the magnetosphere in and around the ring system

Mission Scenario. Due to the rigorous propulsion requirements of a mission of this nature, the nuclear electric propulsion (NEP) system would be used. After an 8.5 year journey to Saturn, the rover would begin spiraling down toward the ring system after orbit insertion utilizing Titan's gravity field. Once the rover has arrived at the ring system, full-scale experiments and observations would begin with the rover travelling above the ring plane on a slowly decreasing spiral orbit.

2.3.10 1995 Uranus Orbiter/Probe

Science Rationale/Objectives. This mission is essentially an uprated version of the flyby/probe mission outlined in NASA's Core Program. It will be a Galileo-type mission providing a reconnaissance of the Uranian satellites and ring system as well as characterizing the Uranian atmosphere. Specific objectives include:

- Determine the size and structure of the magnetosphere;
- Determine the nature and composition of the ring system;
- Determine the composition and geologic history of the satellites;
- Observe the atmospheric dynamics and structure; and
- Determine the internal structure, composition and dynamics of the atmosphere utilizing an atmospheric entry probe.

2.3.11 Uranus Orbiter/Multiprobe (Long Duration)

Science Rationale/Objectives. The scope of this mission is virtually identical to the scope of the proposed Jupiter multiprobe mission with the possible exception that the long-duration probes will be floating at the 100 bar level in order to obtain meaningful meteorological data about the atmosphere. No 1000 bar probe will be used on this mission.

Compelling Science Questions

1. Why is Uranus the only gaseous planet that does not radiate more energy than it receives?
2. Why is Uranus tilted 98° to the ecliptic?
3. What is the composition of its ring system?
4. How large is Uranus' magnetosphere?
5. What effect does its tilt have on its atmospheric dynamics?

Instruments and Expected Results

Orbiter

MULTISPECTRAL IMAGER: observe atmospheric dynamics, ring system and satellites

MAGNETOMETER: measure the intensity and structure of the magnetosphere; observe interaction with solar wind and satellites

PLASMA DETECTOR: determine the composition, energy, and three dimensional distribution of low energy ions and electrons within the magnetosphere

PLASMA WAVE INSTRUMENT: detect electromagnetic waves and analyze wave-particle interactions inside the magnetosphere

ENERGETIC PARTICLE DETECTOR: measure high energy electrons, protons and heavy ions in and around the magnetosphere

UV SPECTROMETER: measure gas and aerosols in the atmosphere

PHOTOPOLARIMETER/RADIOMETER: determine distribution and character of atmospheric particles; compare flux of thermal radiation to incoming solar levels; determine the character of the ring system

NEAR-IR MAPPING SPECTROMETER: determine satellite surface composition as well as the composition of the atmosphere of Uranus

DUST DETECTOR: measure mass, volume, size and charge of particles in the magnetosphere

Multiprobe

NEPHELOMETER: determine presence of cloud layering by measuring vertical distribution of particles

NEUTRAL MASS SPECTROMETER: determine chemical composition of atmosphere

ATMOSPHERIC STRUCTURE INSTRUMENT: measure the temperature, pressure, molecular weight and density of the atmosphere

HELIUM ABUNDANCE DETECTOR: measure the ratio of H/He in the atmosphere

LIGHTNING AND RADIO EMISSION DETECTOR/ENERGETIC PARTICLE INSTRUMENT: measure energetic particles in inner magnetosphere; determine presence of lightning in atmosphere

GAS CHROMATOGRAPH: measure the abundance of organic and inorganic compounds at different altitudes within the atmosphere

NET FLUX RADIOMETER: determine the level of solar radiation to thermal emission of the planet within the atmosphere; determine mixing ratios of various atmospheric constituents

Mission Scenario. The orbiter begins observation of Uranus 60 days prior to orbit insertion with its imaging system. Fifty days prior to encounter, the probe is released for free-fall into the atmosphere of Uranus. Fifteen days before insertion, the orbiter begins searching for a bow shock of the magnetosphere. The orbiter enters orbit at the same time that the probe enters the atmosphere and acts as a communications link between the probe and the Earth. The probe will take measurements down as deep as 100 bars in order to obtain readings from the water cloud layers. Once the probe mission has been completed, the orbiter will continue to survey the Uranian system.

2.3.12 Neptune Orbiter/Dual Probe

Science Rationale/Objectives. The Neptune Orbiter/Dual Probe mission will send one probe into Neptune's atmosphere and one into Triton's. Specific objectives include:

- Determine the existence of a magnetosphere around Neptune;
- Determine the composition, structure and dynamics of the

atmosphere of Neptune;

- Determine the nature and extent of the recently discovered ring system;
- Map the surface features of the satellites;
- Determine the nature and composition of Triton's atmosphere;
- Determine the nature of Triton's surface; and
- If Triton probe survives impact, accelerometer will continue to monitor for seismic activity.

Compelling Science Questions

1. Does Neptune have a magnetosphere?
2. Does Triton have seas of liquid nitrogen?
3. What is the size and extent of the newly discovered ring system of Neptune?
4. Why does Triton revolve in a retrograde motion about Neptune?
5. How does Triton's atmosphere compare with Titan's?
6. Why does Triton possess an atmosphere at all?

Instruments and Expected Results

Orbiter and Neptune Probe instrumentation are identical to that proposed for Uranus.

Triton Probe

NEPHELOMETER: measure the size of the cloud particles and determine the location of layers in the atmosphere

NEUTRAL MASS SPECTROMETER: measure the chemical composition of the atmosphere

ATMOSPHERIC STRUCTURE INSTRUMENT: measure the temperature, pressure, density and molecular weight of the atmosphere

IR RADIOMETER: measure the vertical distribution of IR radiation in the atmosphere; detect cloud layers

DESCENT IMAGER: provide images of the surface prior to impact; measure spectral radiance of the atmosphere

GAS CHROMATOGRAPH: measure trace constituents in the atmosphere such as noble gases and organic compounds

Possible additions if mass constraints allow:

Pre-entry science package

- RETARDING POTENTIAL ANALYZER: measure the thermal properties and structure of upper atmosphere
- ELECTRON TEMPERATURE PROBE: measure the densities of electrons and ions and electron temperature in the ionosphere
- ION MASS SPECTROMETER: determine the composition of the ionosphere by measuring ionized gases
- NEUTRAL MASS SPECTROMETER: determine the density and abundance of neutral gas species in the atmosphere

Mission Scenario. Observations of Neptune will begin approximately 60 days prior to orbit insertion, allowing atmospheric activity and any auroral activity to be monitored. The Neptune probe is released approximately 50 days prior to orbit insertion to arrive at Neptune's atmosphere at the same time as the orbiter. The orbiter begins searching for Neptune's bow shock 10-15 days before encounter. Several days before the next Triton encounter, the Triton probe is released to arrive at the same time as the orbiter. The Neptune probe will conduct measurements down to the 100 bar level. The Triton probe, however, will conduct measurements of atmospheric properties until impact on the surface. If the probe survives impact, the accelerometer will continue to operate, monitoring any seismic activity that may be present. After the probes have completed their observations, the orbiter will begin intensive observation of the satellites of Neptune, the atmospheres of Neptune and Triton, and Neptune's ring system (if one exists).

2.3.13 Pluto Orbiter/Lander and Charon Lander

Science Rationale/Objectives. This mission will characterize the Pluto/Charon system as well as measure the farthest known boundary of the interplanetary environment. Specific objectives include:

- Determine the presence of a magnetic field;
- Conduct general planetology of both bodies;
- Measure the interplanetary environment at Pluto's orbit;
- Determine the surface conditions and composition by utilizing separate hard landers for Pluto and Charon; and
- Determine the interaction between Pluto and Charon (i.e. tidal effects, etc.).

Compelling Science Questions

1. Does Pluto have an atmosphere?
2. What is the surface composition of Pluto and Charon ?
3. What is the character of the interplanetary medium at Pluto's orbit?
4. Is there an as-yet-undiscovered body beyond the orbit of Pluto?
5. What is Pluto's bulk composition?

Instruments and Expected Results

Orbiter

MULTISPECTRAL IMAGER: characterize the surfaces of Pluto and Charon

MAGNETOMETER: search for a magnetosphere and any interaction with the diffuse solar wind

PLASMA DETECTOR: measure the low energy particles within the planetary environment

ENERGETIC PARTICLE DETECTOR: measure high energy electrons, protons and heavy ions in the planetary environment

UV-VIS-IR MAPPING REFLECTANCE SPECTROMETER: map the mineralogical and elemental composition of Pluto and Charon

GAMMA-RAY SPECTROMETER: map abundances of radioactive elements on surfaces

RADAR ALTIMETER: provide surface topography measurements

MULTICHANNEL MICROWAVE RADIOMETER: measure the thermophysical properties of Pluto and Charon including heat flow, temperature porosity, etc.

DUST DETECTOR: detect the presence of interplanetary dust and measure its size, shape and charge

PLASMA WAVE INSTRUMENT: measure wave-particle interaction in the planetary environment

Pluto and Charon Lander

3-AXIS SEISMOMETER: monitor for any seismic activity

MULTISPECTRAL IMAGER: provide photos of surface morphology and character

ALPHA-PROTON BACKSCATTER/X-RAY FLUORESCENCE SPECTROMETER: determine elemental composition at the surface

MAGNETOMETER: measure the presence of any local magnetic field and any fluxes in the field

TEMPERATURE SENSOR: determine heat flow values

Mission Scenario. While the spacecraft is in orbit about Pluto, low resolution imaging and mapping will commence in order to select a suitable landing site for a small lander. Once a site is chosen, the lander will separate from the orbiter. The Charon lander will then be targeted and launched before close encounter with Charon. Measurements at the surfaces will then proceed in conjunction with those made in orbit.

3. VEHICLE HARDWARE CANDIDATES AND ASSEMBLED SPACECRAFT MASS

In this section, the hardware elements needed to support the candidate missions are described and their technology readiness is assessed. The description of each hardware item includes general capabilities and heritage from other vehicles or programs. The technology assessment rates a particular hardware element as having a high, moderate or low level of readiness based on the following criteria.

High - Hardware item is an exact repeat or minor modification of a design used for a previously flown or soon to be flown mission. An example of this would be a Galileo-type atmospheric entry probe which has been designed, built and tested.

Moderate - Hardware item is a concept which uses proven technology from other non-related programs but which has never been constructed and tested itself. An example of this would be an aerocapture vehicle which uses reentry systems, guidance and control systems, and aerodynamics which are all well-understood but a vehicle of the size and complexity under discussion has never been built.

Low - Hardware item is a concept only with little or no previous hardware having been built for either related or unrelated areas. The NEP stage would be an example of this category due to the lack of any previous experience in designing a space-rated nuclear reactor with the associated power processing units and ion bombardment thrusters.

A mass estimate is included for each of the hardware elements to be discussed. This estimate is to the level of detail allowed by the technology readiness level. These mass estimates are then assembled as necessary to provide an overall estimate of the vehicle mass required to accomplish a specific mission.

3.1 Vehicle Hardware Candidates

3.1.1 Spacecraft Support Bus

Each of the candidate missions discussed in previous sections will require a suitable spacecraft bus to provide essential engineering support systems, such as power, thermal control, communications and pointing, and to act as a transport vehicle for any deployed payloads. The majority of the missions discussed can be supported by one of two generic vehicles - the Mariner Mark II or the spacecraft bus designed to operate with the NEP stage. Some of the candidate missions are either so specialized, as in the case of the Jupiter Inner Magnetosphere mission, or hardware needs are so specific, such as the probe carrier for the Jupiter Multiprobe and Titan Buoyant Station missions, that use of generic vehicles would be inappropriate. In these special cases, a mission-unique bus is hypothesized to carry out its mission in an efficient manner. Each of these vehicles, the generic as well as special purpose, are discussed in turn in the sections which follow.

The **Mariner Mark II** concept was developed as a result of recommendations made by the SSEC for the exploration of comets, Mainbelt asteroids and the outer planets (Ref. 16). The more demanding environmental constraints and communication distances involved for these missions disallowed the use of a modified Earth-orbiting vehicle such as that used in the Planetary Observer program. However, the SSEC recommended that the unit cost for these missions be reduced below the level of current outer planet missions while maintaining the same data quality obtained from Voyager and Galileo. This would be accomplished by constraining the missions to accept a common design for the engineering segment of the spacecraft. Cost reductions from heritage would also be obtained through the use of hardware and designs from previous missions. Examples include the 10-bay bus design from Voyager, the high gain antennas from either Voyager or Galileo, and the RTG designs (MHW and GPHS) used for Voyager and Galileo. The overall bus design would be flexible enough to function on a flyby or orbiter mission and would be capable of delivering Galileo-type entry probes or other small deployable vehicles (Figure 3-1). Any new technology which would be incorporated into these missions would be used to reduce mission cost rather than improve the science return. It has been assumed for this study that the basic features of this bus have been fixed and that any development effort for a particular mission will go into the unusual features of that mission. Based on this assumption, Table 3-1 provides a basic mass statement for the Mariner Mark II bus, although the total may vary slightly from mission to mission due to these "unusual features".

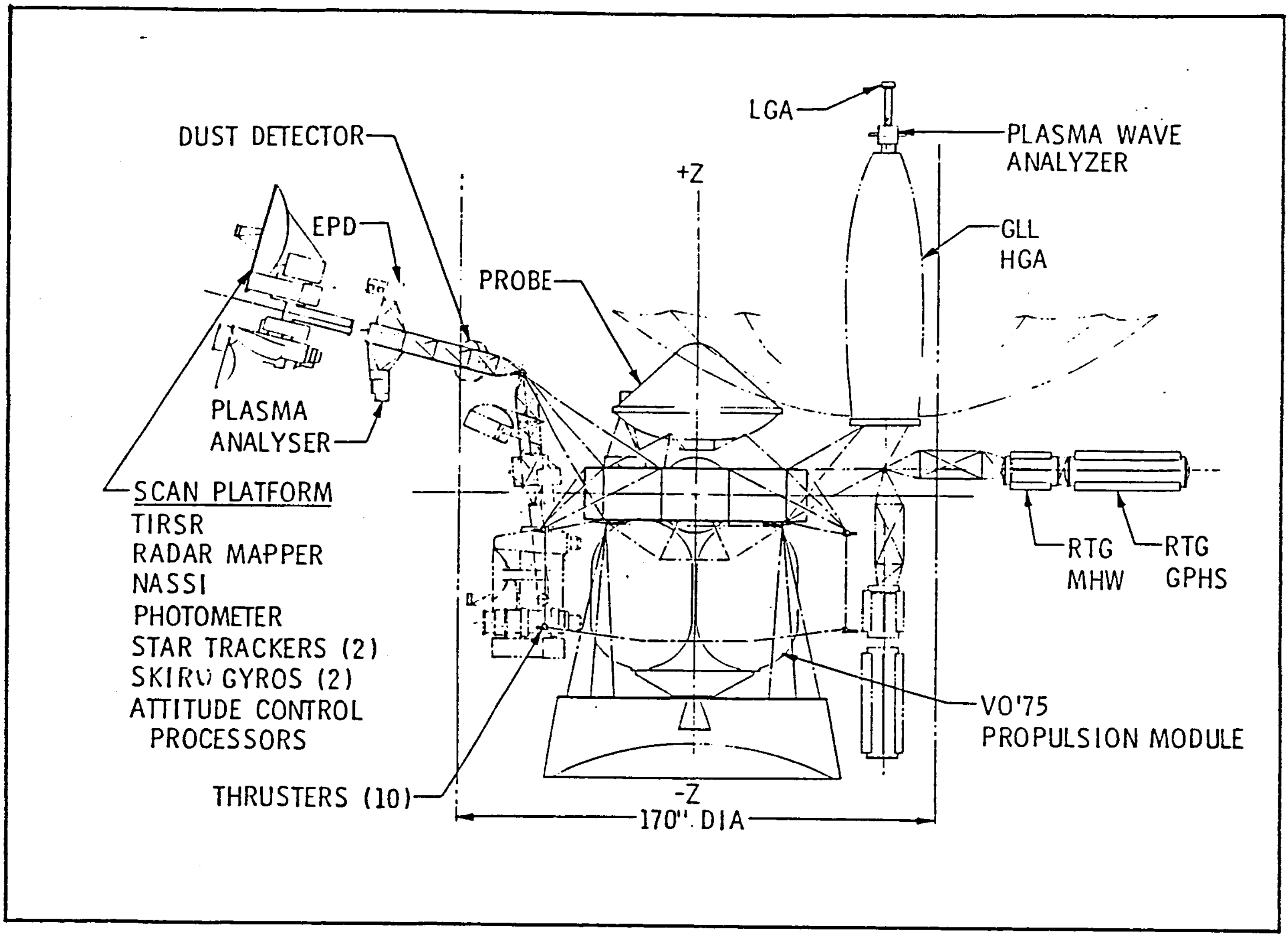

Figure 3-1. Sample Mariner Mark II Bus - Outer Planet Orbiter-Probe Configuration (JPL Drawing)

Table 3-1

MARINER MARK II BUS MASS BREAKDOWN

Structure	172 kg
Devices	26
Thermal Control	76
Cabling	52
Pyro	4
AACS	81
Power	169
Antenna	36
RFS	20
CDH	24
DSS	9
Contingency	76
Total	745 kg

Figure 3-1 illustrates a typical configuration for the Mariner Mark II on an outer planets-type mission. This figure also shows a deployable entry probe and propellant tankage, neither of which was included in Table 3-1 since each mission will have unique requirements regarding these subsystems. Given the assumptions made above, this vehicle will have a high level of technology readiness by the time any of the candidate missions would be flown.

The **NEP Spacecraft Bus** will provide a set of functions similar to the Mariner Mark II but will be configured for use with the NEP stage. Use of a nuclear power source for this type of spacecraft requires the selection of components which can operate in a nuclear radiation environment. This could involve hardening of electronic components and/or shielding of selected systems by a suitable inert material. The actual configuration and mass estimate for this type of bus will continue to evolve as the NEP stage evolves. However, a recent study of the Saturn Ring Rover mission (Ref. 17) has developed one concept for this bus which will be used here.

Figure 3-2 illustrates the bus for this vehicle with various science platforms and the high gain antenna deployed. The engineering support subsystems of concern here will be located within the 2.0x1.8x0.5m bus section

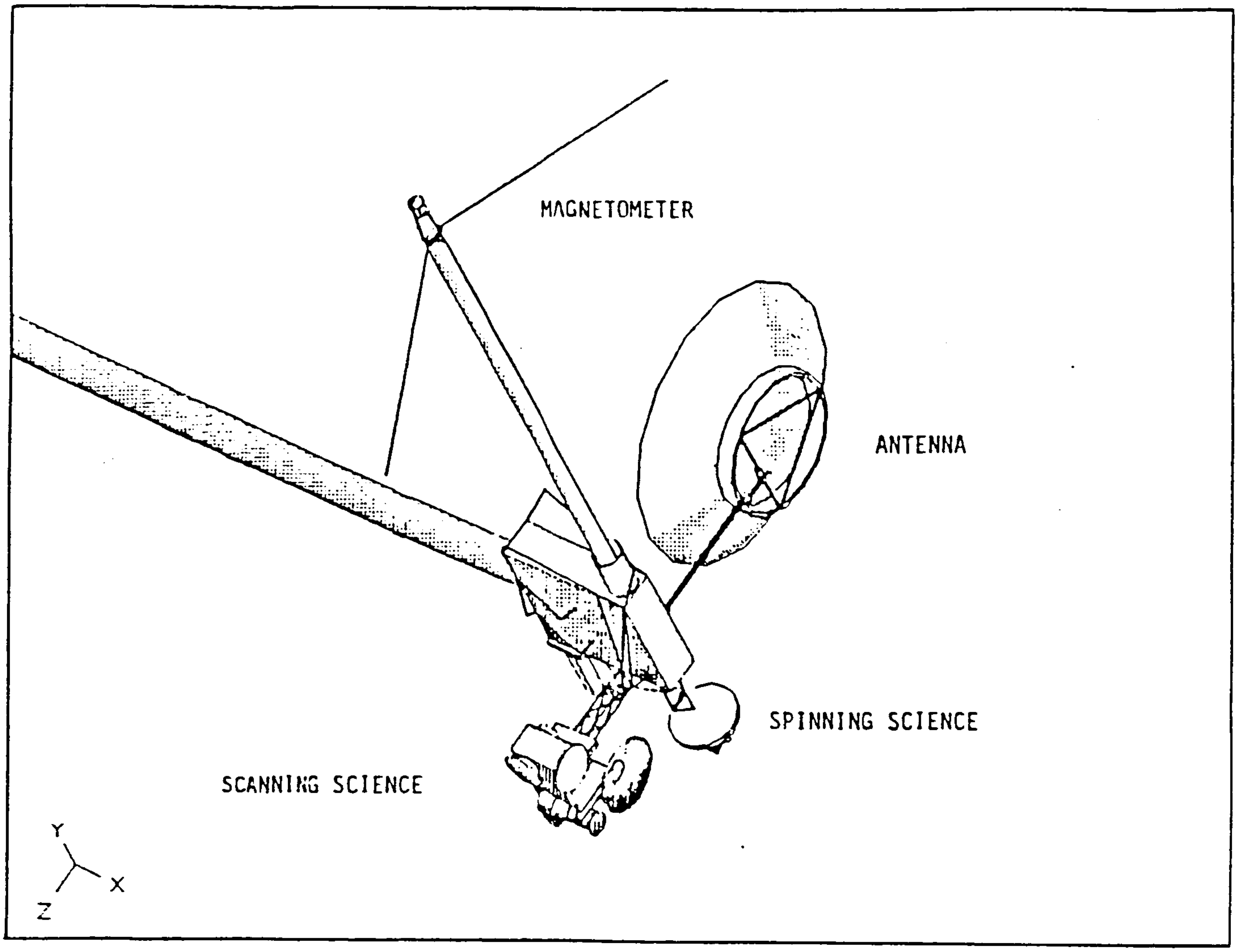

Figure 3-2. Sample NEP Bus (JPL Drawing)

which also serves as the mounting structure for the appendages. Table 3-2 provides an estimate of the mass for this bus.

Table 3-2

NEP SPACECRAFT BUS MASS BREAKDOWN

Structure and Devices	325 kg
Thermal, Cabling and Pyro	111
AACS	180
Power	200
Antenna	50
RFS	150
Data System	48
Boom	120
Contingency	175
Total	1359 kg

Figure 3-3 indicates the position of the spacecraft bus relative to the NEP reactor and electric propulsion modules. In this configuration the bus is located 39 meters from the electric propulsion module. This allows the mercury propellant tank to be used as a shadow shield for the bus. Power is supplied to the spacecraft by means of a cable from the power processing units in the propulsion module.

Given the uncertain nature of the final NEP stage configuration and the dependence of the spacecraft bus on this configuration, the current technology status of this vehicle must be rated as low to moderate.

Three types of **Special Mission Spacecraft Buses** will be required for several of the candidate missions. These include: (1) the bus for the Jupiter Inner Magnetosphere mission vehicle; (2) the bus for the Jupiter Deep Probe/Multiprobe; and (3) a Titan probe carrier. None of these buses requires the full capability of the Mariner Mark II or the NEP bus because of the more focused or limited nature of the task each will carry out.

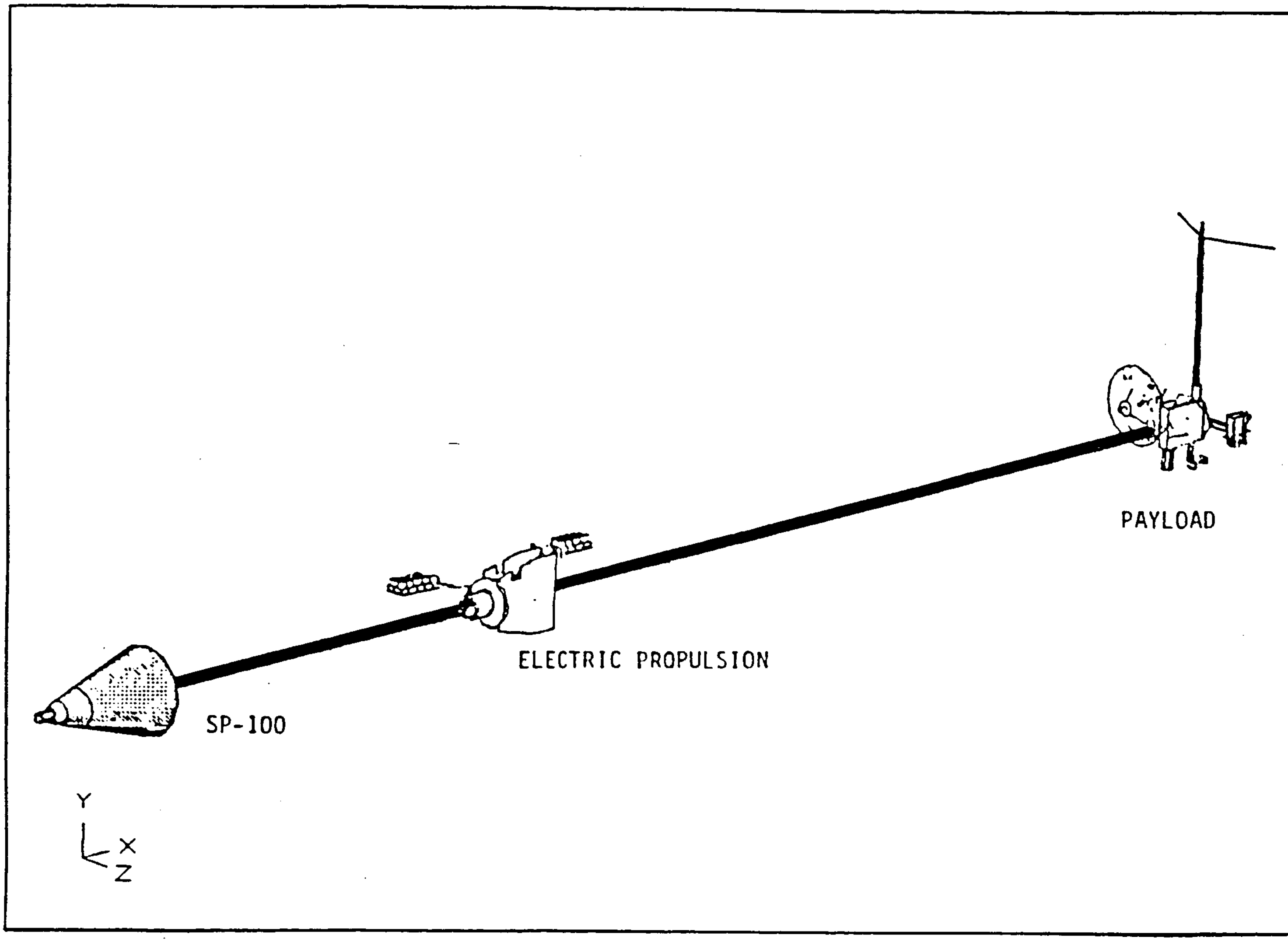

Figure 3-3. NEP Bus Location Relative to Power Source and Propulsion Module (JPL Drawing)

The Jupiter Inner Magnetosphere mission presents a unique set of engineering requirements which may be difficult to meet for mission completion. This relates specifically to the radiation environment in which the vehicle will be operating. A spacecraft specifically hardened to survive in this environment will use techniques similar to those proposed for the NEP vehicle, such as hardening of sensitive electronics and/or shielding of selected systems by a suitable inert material. The overall vehicle configuration will be dictated by several factors. The science instrument complement with a fields and particles focus indicates the desirability of a spin-stabilized spacecraft. In addition, the instrument complement is small compared to other candidate missions. Both of these factors argue against using a bus of the size and capacity of either the Mariner Mark II or the NEP vehicle. Rather, an RTG-powered, spin-stabilized vehicle similar to that depicted in Figure 3-4 would best serve the objectives of this mission. A mass breakdown of this vehicle is presented in Table 3-3.

Table 3-3

JUPITER INNER MAGNETOSPHERE/POLAR ORBITER SPACECRAFT BUS MASS BREAKDOWN

Structure and Devices	139 kg
Thermal, Cabling and Pyro	52
AACS	20
Telecommunications	30
Antennas	8
Command and Data Handling	50
RTG	77
Power Support	35
Radiation Shielding	20
Contingency	50
Total	481 kg

With the exception of the necessary radiation hardening, all subsystems of this concept would be drawn from previously tested designs. The need for new radiation hardening combined with previous subsystem design experience implies a moderate level of technology readiness.

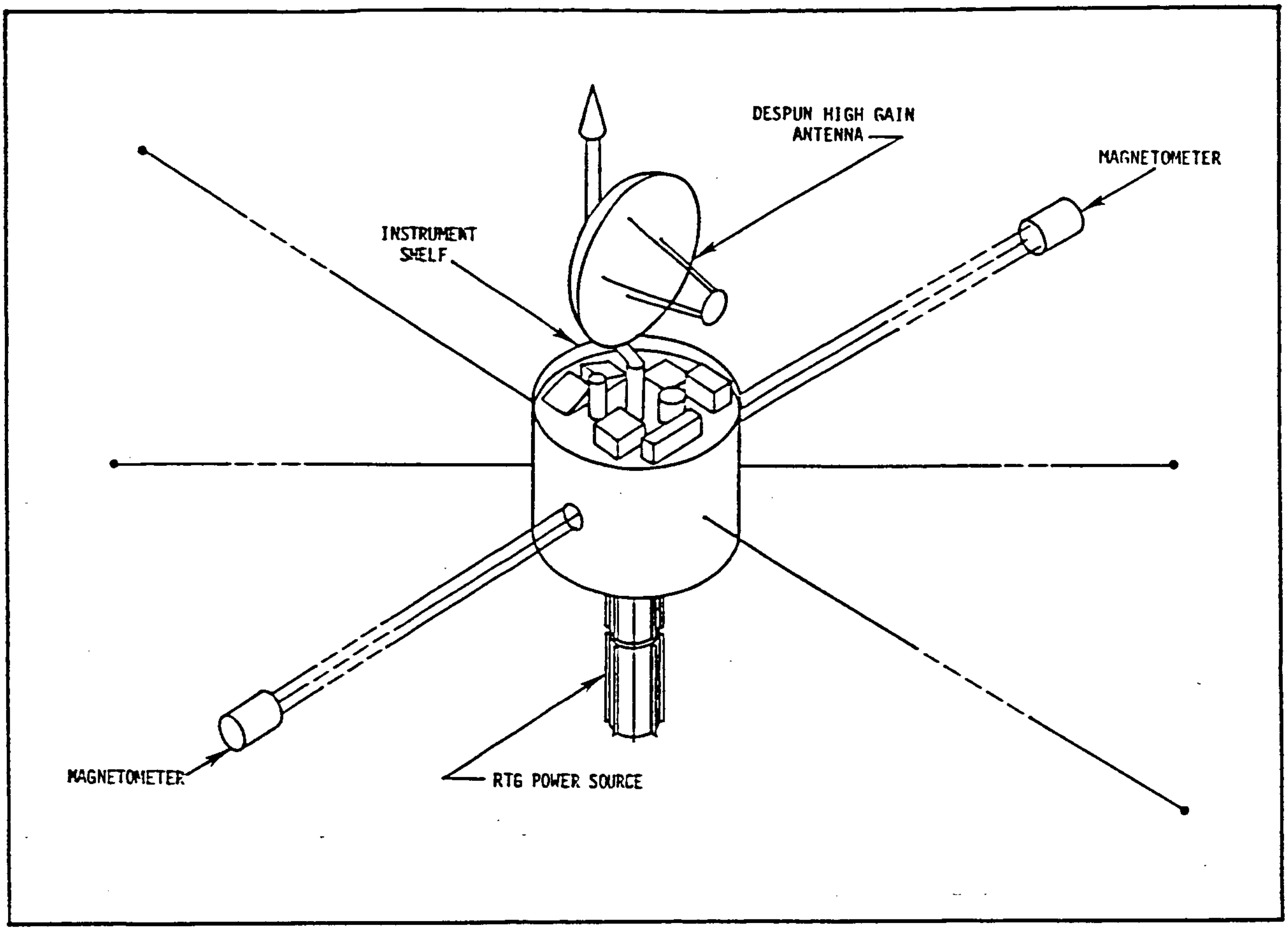

Figure 3-4. Representative Configuration for the Jupiter Inner Magnetosphere Spacecraft

The second type of special mission spacecraft bus will be used to support atmospheric entry probes while en route to the target body. This type of vehicle will be necessary for one of the Titan exploration options and the Jupiter Deep Probe/Multiprobe mission. The Titan probe carrier would be used in conjunction with a separate orbiter vehicle. For this particular mission, the entry probes would be released on approach to Titan and the carrier vehicle would continue past Titan on a flyby trajectory. A vehicle concept proposed by Hughes Aircraft Corporation (Ref. 18) as a Galileo probe carrier would be well suited to this type of mission. This vehicle configuration is illustrated in Figure 3-5 with a mass breakdown contained in Table 3-4. Suitable adapters would be required for this vehicle depending on the exact mission scenario as well as the number and type of probes to be carried.

Table 3-4

TITAN PROBE CARRIER MASS BREAKDOWN

Structure and Devices................	191 kg
Thermal, Cabling and Pyro............	60
AACS.................................	17
Telecommunications...................	21
Command and Data Handling............	30
Power Source and Processing..........	94
Contingency	41
Total................................	454 kg

The technology readiness for this type of vehicle is high due to previous design experience with all subsystems involved.

The Jupiter Deep Probe/Multiprobe bus will function in much the same manner as the Titan probe carrier in that it will provide general support functions during the dormant cruise phase. As such, the general configuration could be similar to the Titan probe carrier. However, several significant alterations will be required. The scenario for this mission requires the bus to enter orbit at Jupiter and act as a communications relay for each probe deployment. This necessitates a different communication system and data system which can track the probe and record the returned data for later

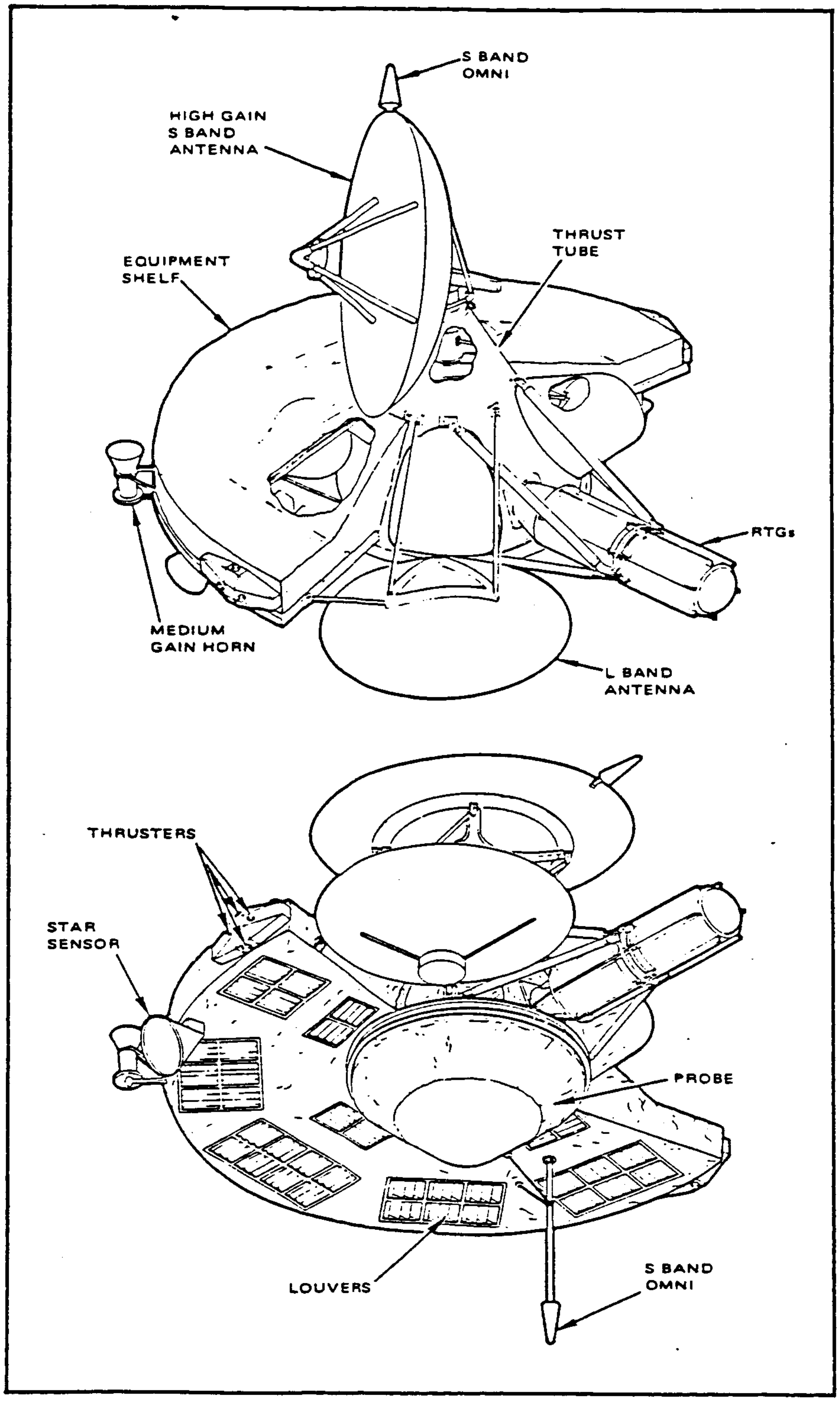

Figure 3-5. Sample Configuration for a Probe Carrier Vehicle (Hughes Drawing)

playback to Earth. The larger size and mass of the probes for this mission will require that the bus have an enhanced structure to accommodate larger launch and orbit insertion loads. The orbit insertion itself will require a significant change in the type and capacity of the on-board propulsion system compared with the Titan vehicle. A probe carrier of this kind is estimated to have the mass characteristics listed in Table 3-5.

Table 3-5

JUPITER DEEP PROBE/MULTIPROBE SUPPORT BUS MASS BREAKDOWN

Structure and Devices	252 kg
Thermal, Cabling and Pyro	60
AACS	39
Telecommunications	41
Antennas	20
Command and Data Handling	53
Power Source and Processing	112
Contingency	58
Total	635 kg

As with the Titan probe carrier, the technology readiness of this vehicle is high.

3.1.2 Atmospheric Entry Probes

Many of the candidate missions in this study will utilize atmospheric entry probes for a major segment of each mission. All probes under consideration here would have the same basic configuration and deployment sequence. However, the specific dimensions and certain configuration features will be dictated by the mission and the target body. The general configuration (Ref. 31) is illustrated in Figure 3-6 and will consist of a probe module and its deceleration module. The deceleration module itself is made up of a conical heatshield forebody and a spherical segment aft cover. A typical deployment sequence of a probe of this type is shown in Figure 3-7.

Within the candidate mission set, three types of probe modules will be required to cover three different atmospheric pressure levels. For atmos-

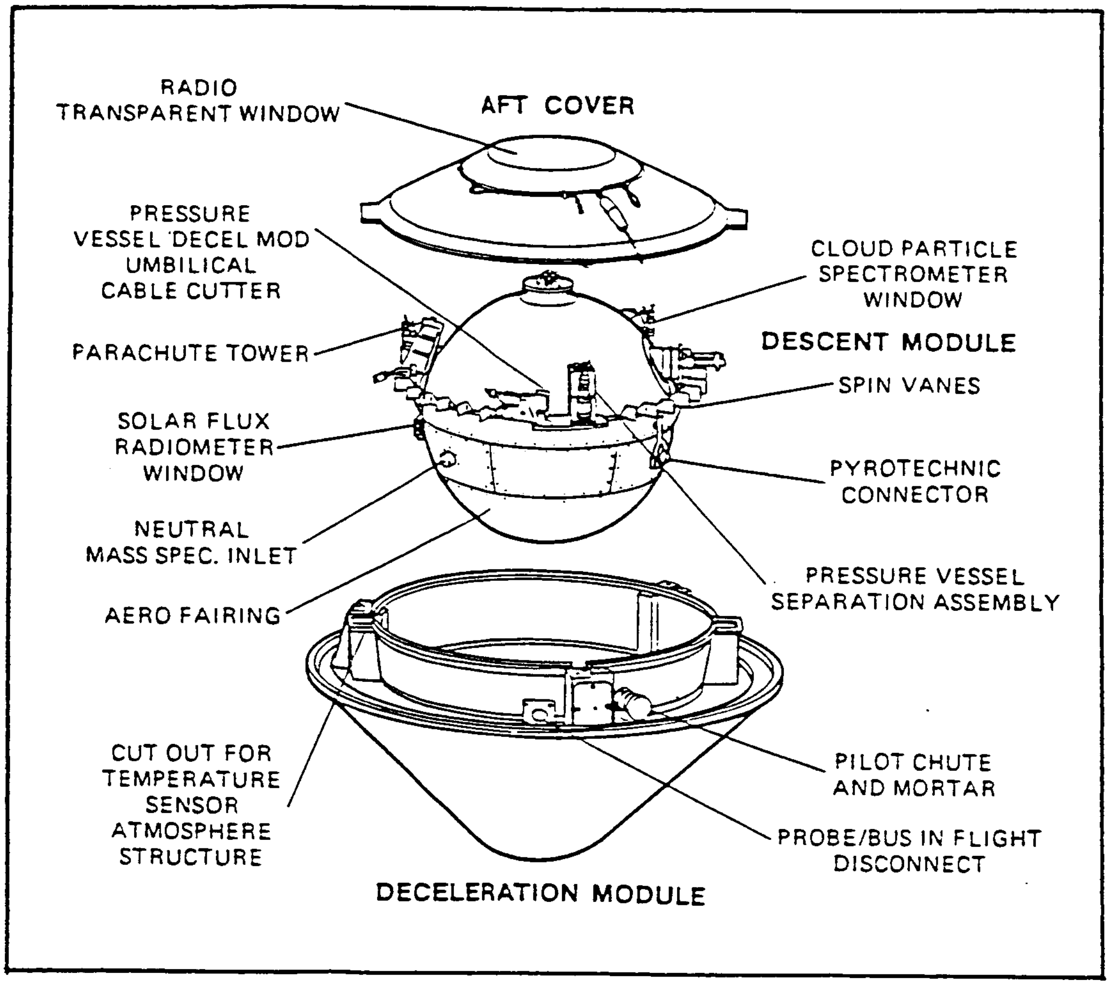

Figure 3-6. Sample Atmospheric Probe Vehicle - Pioneer Venus Large Probe Configuration (NASA/Ames Drawing)

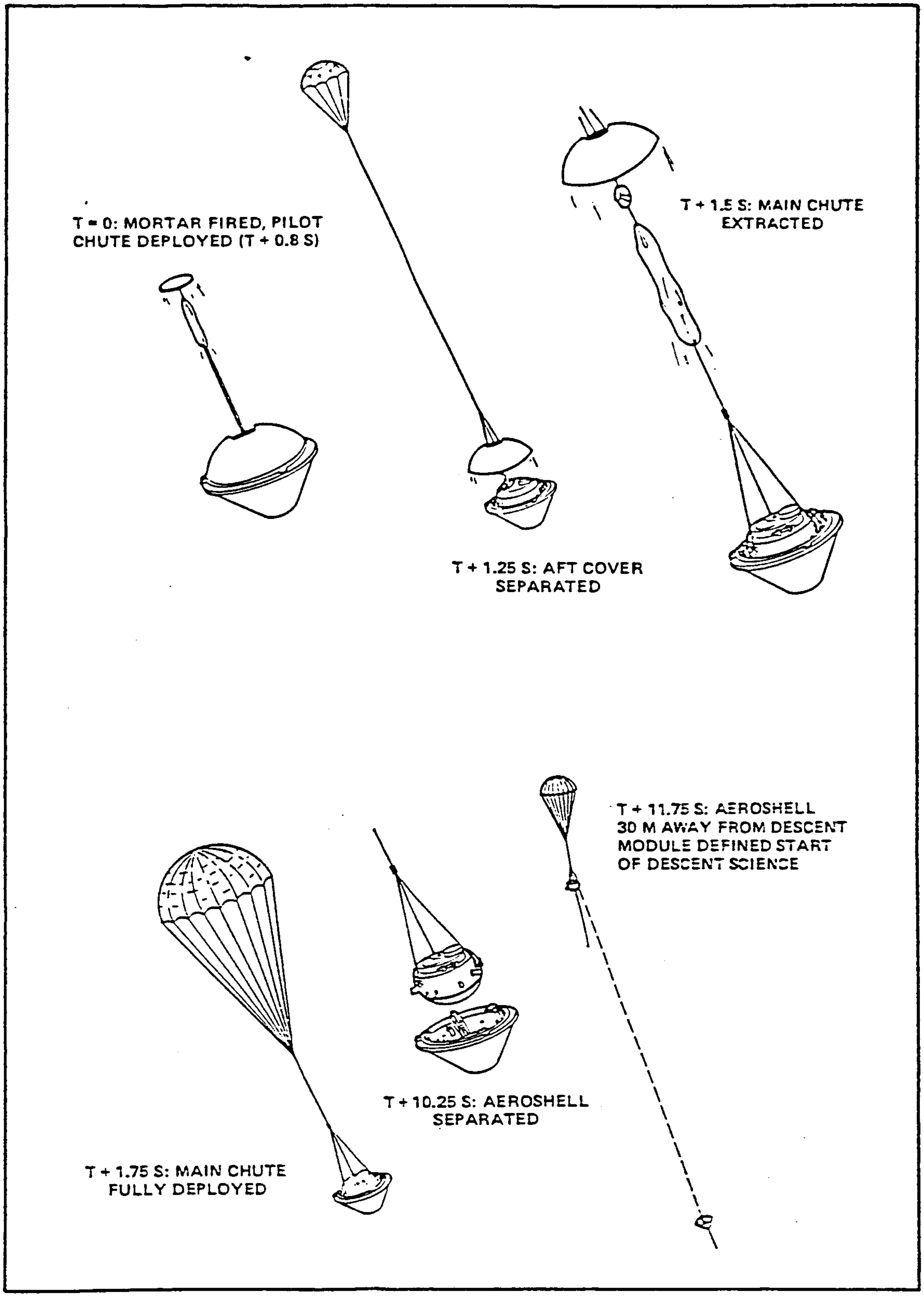

Figure 3-7. Atmospheric Entry Probe Deployment Sequence (Hughes Drawing)

pheric penetrations to depths not exceeding 20 bars, a Galileo-type vented probe will be used (Ref. 18). Pressures between 20 and 100 bars will require a sealed pressure vessel of higher strength and thus of greater mass. Technology and construction techniques for this type of vehicle have been developed as part of the Pioneer Venus program (Ref. 19). The final probe type will be used for pressures up to 1000 bars. Again, a sealed pressure vessel will be used but there exists no previous experience from which to build. Titanium and Inconel were both investigated as possible pressure vessel material using typical wall thickness algorithms for spherical shells. Table 3-6 lists a comparison of probe module mass breakdowns for the Galileo, Pioneer Venus and two types of 1000 bar probes (including the estimated pressure vessel mass).

Table 3-6

PROBE MODULE MASS BREAKDOWN

	Galileo (20 bar)	Pioneer Venus (100 bar)	Titanium (1000 bar)	Inconel (1000 bar)
Structure and Devices	35	143	264	102
Thermal, Cabling and Pyro	9	9	15	15
Telecommunications	11	11	11	11
Antennas	2	2	2	2
Command and Data Handling	17	17	17	17
Power Source and Processing	13	13	20	20
Relay Probe	0	0	30	30
Contingency	-	-	36	18
Total	87 kg	195 kg	395 kg	215 kg

This information shows that Inconel offers a significant advantage over titanium for a lower mass pressure vessel which in turn lowers the mass requirement for the deceleration module. As a result the single deep probe mission will assume an Inconel pressure vessel.

With the exception of this Inconel pressure vessel, all subsystems have been built and tested for other missions giving these vehicles a high technology readiness level.

An additional feature which would make these probes much more attractive would be the ability to remain aloft, at survivable pressure levels, for extended periods of time. The first concept which comes to mind in this context is a balloon. The French Space Agency has used this concept successfully at Venus (Ref. 20) and a previous study indicates that a similar application at Titan will work as well (Ref. 21). Figures 3-8 and 3-9 show how the Titan probe system would be stored for entry and subsequently deployed.

However, the application of balloons at the gas giants will not work for small payloads. A previous study (Ref. 22) along with the present investigation indicate that a buoyant gas is not feasible since the atmosphere is of approximately the same molecular weight and a "hot air" configuration applied at Jupiter will require excessive amounts of power, as illustrated in Figure 3-10, to provide the necessary thermal differential. Figure 3-11 shows that insulating the fabric material of this type of balloon also fails to help generate a sufficient thermal gradient.

One possible alternative is to use a wind gradient to support the probe. With a sufficiently long tether (several tens of kilometers in length) and a favorable wind gradient, an aerodynamic structure such as a kite or an inflatable airfoil will provide the necessary lift. Lack of any direct knowledge regarding the size of the wind gradient and possible large scale turbulents makes the feasibility of this concept uncertain. The Galileo probe may provide useful insight into these questions for Jupiter and possibly the other gas giants. Despite this uncertainty, this concept is worth further examination.

The technology for kites, airfoils and long tethers has been demonstrated for Earth-based systems but not in the combination desired here. This, in addition to uncertainty in actual atmospheric conditions and its effects on this type of system indicate that the technology readiness is low to moderate.

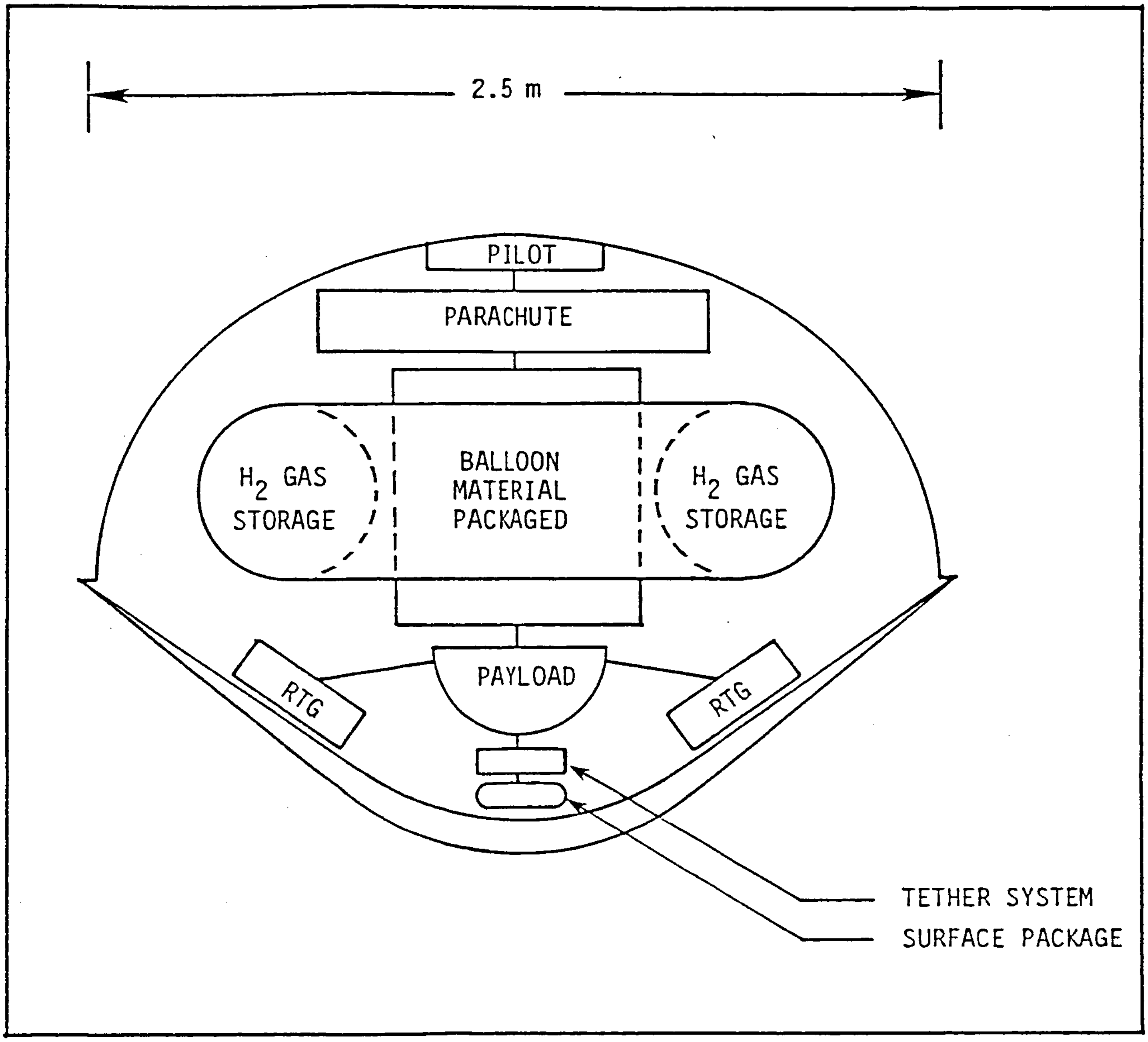

Figure 3-8. Large Buoyant Station Package in a Titan Entry Probe

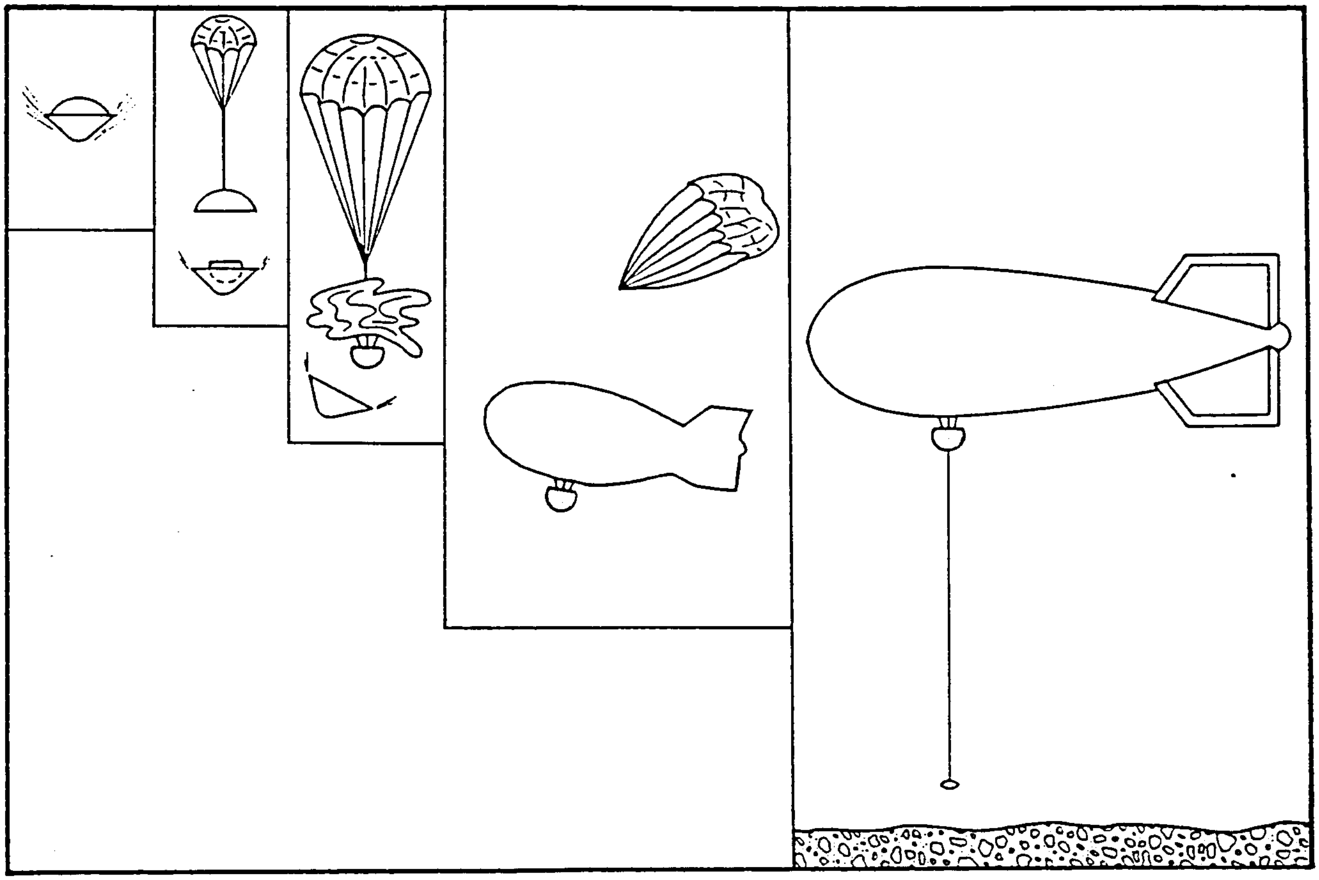

Figure 3-9. Deployment Sequence for a Typical Buoyant Station Vehicle

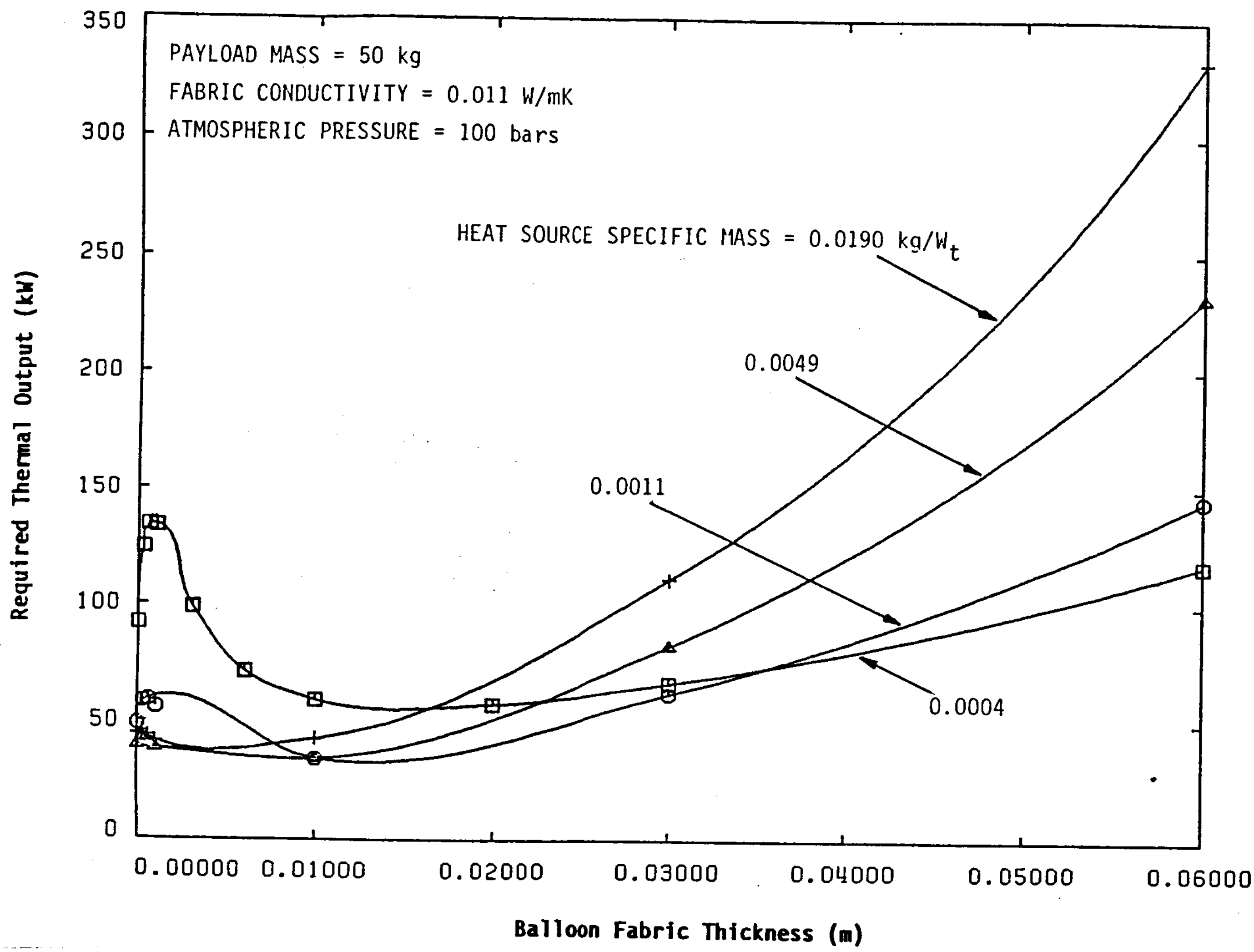

Figure 3-10. Thermal Power Required to Maintain a Jupiter "Hot Air" Balloon at a Pressure Level of 100 Bars

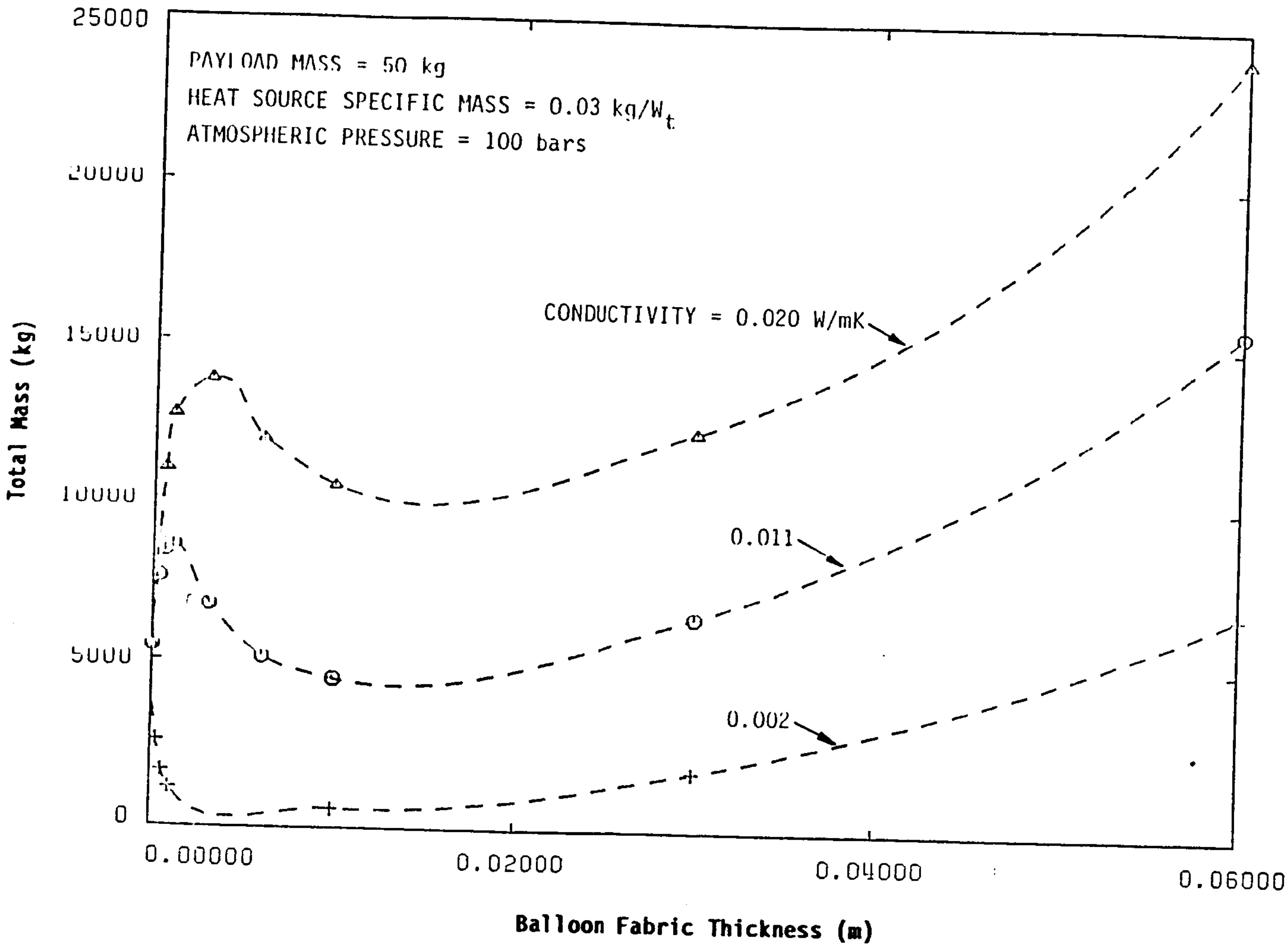

Figure 3-11. Total Jupiter "Hot Air" Balloon Mass Required to Stay Aloft at a Pressure Level of 100 bars for Various Fabric Insulation Values

3.1.3 Landers

Requirements for two types of surface landers have been identified from the candidate mission scenarios. A soft landing capability will be necessary for the Europa Orbiter/Lander mission. The Pluto/Charon reconnaissance orbiter mission also calls for instruments to be placed on the surface but a vehicle with capabilities similar to those needed at Europa is not required. A simpler lander capable of surviving higher impact velocities will be sufficient.

An interplanetary spacecraft with a soft landing capability has been developed and used as part of the Viking mission to Mars (Ref. 23). The same basic design, as illustrated in Figure 3-12, will be suitable for the Europa mission if several alterations are incorporated for this particular application. The lack of an atmosphere removes the requirement for an aerodeceleration shield and parachute but the bioshell will still be needed for planetary quarantine purposes. The lander will communicate with Earth through the orbiter spacecraft reducing the mass and power requirements for the telecommunication subsystem. However, a more sophisticated pointing and tracking system to maintain a communications lock with the orbiter may negate these gains. Among the other internal subsystems, the elimination of the biology experiment would provide sufficient space for other science instruments including an expanded geophysics instrument set. The sampler system would be retained to assist in the geophysics experiments but this device may require enhancements to adjust to what is expected to be a surface composed of icy materials. For a similar reason, the bottom surface of the main body and landing leg pads must be suitably insulated to reduce the thermal energy transferred to the surface, possibly corrupting its original state and reducing its scientific value. Finally, as noted for the Jupiter Inner Magnetosphere mission vehicle, radiation exposure in Jupiter space, even at the orbit radius of Europa, will be significant. The lander electronics will thus require hardening or shielding to attain a reasonable lifetime for the vehicle. Assuming these alterations, Table 3-7 shows the resulting mass breakdown for the soft lander.

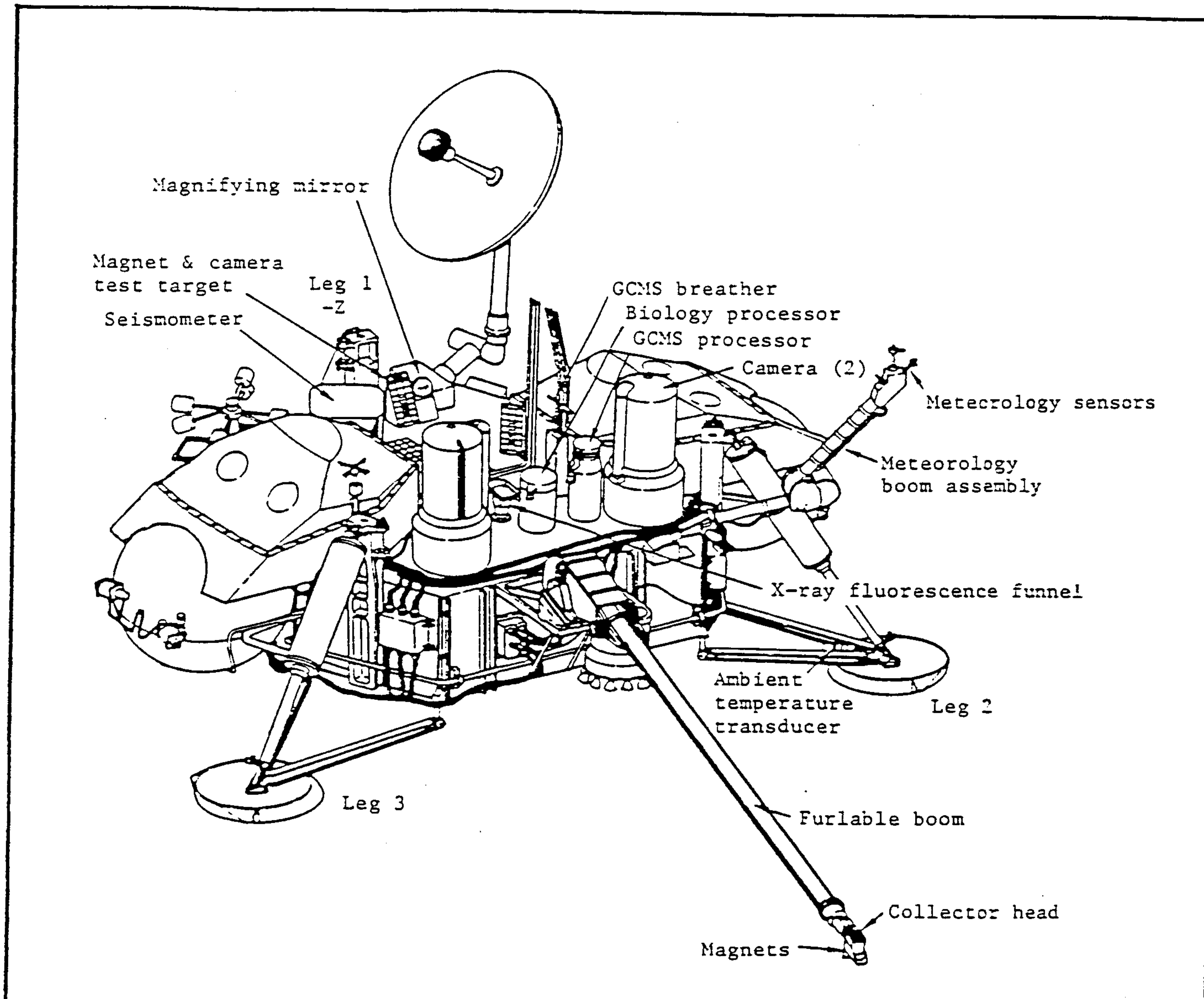

Figure 3-12. Viking Lander Configuration (NASA/Langley Drawing)

Table 3-7

SOFT LANDER SPACECRAFT MASS BREAKDOWN

Structure and Devices	117 kg
Thermal, Cabling and Pyro	58
AACS	74
Radiation Shielding	34
Telecommunications	22
Antennas	10
Command and Data Handling	9
Power Source and Processing	118
Contingency	57
Total	499 kg

Since these vehicle systems are assumed to be derived from the Viking spacecraft but with a requirement for new or revised hardware subsystems, the technology readiness for this vehicle is rated as moderate to high.

The second surface exploration lander identified from the candidate missions has a capability requirement which lies between the soft lander just discussed and a penetrator vehicle, which will be described in the next section. Several vehicle studies have been conducted for a hard lander which could, at a minimum, carry out the same type of exploration as a penetrator while retaining the same vehicle mass as the penetrator (Refs. 24 and 25). One of these concepts is illustrated in Figure 3-13 with the associated mass breakdown listed in Table 3-8.

Table 3-8

HARD LANDER VEHICLE MASS BREAKDOWN

Structure and Devices	11.0 kg
Thermal Cabling and Pyro	4.1
AACS	3.0
Telecommunications	2.0
Antennas	0.5
Command and Data Handling	2.0
Power Source and Processing	6.0
Contingency	3.0
Total	31.6 kg

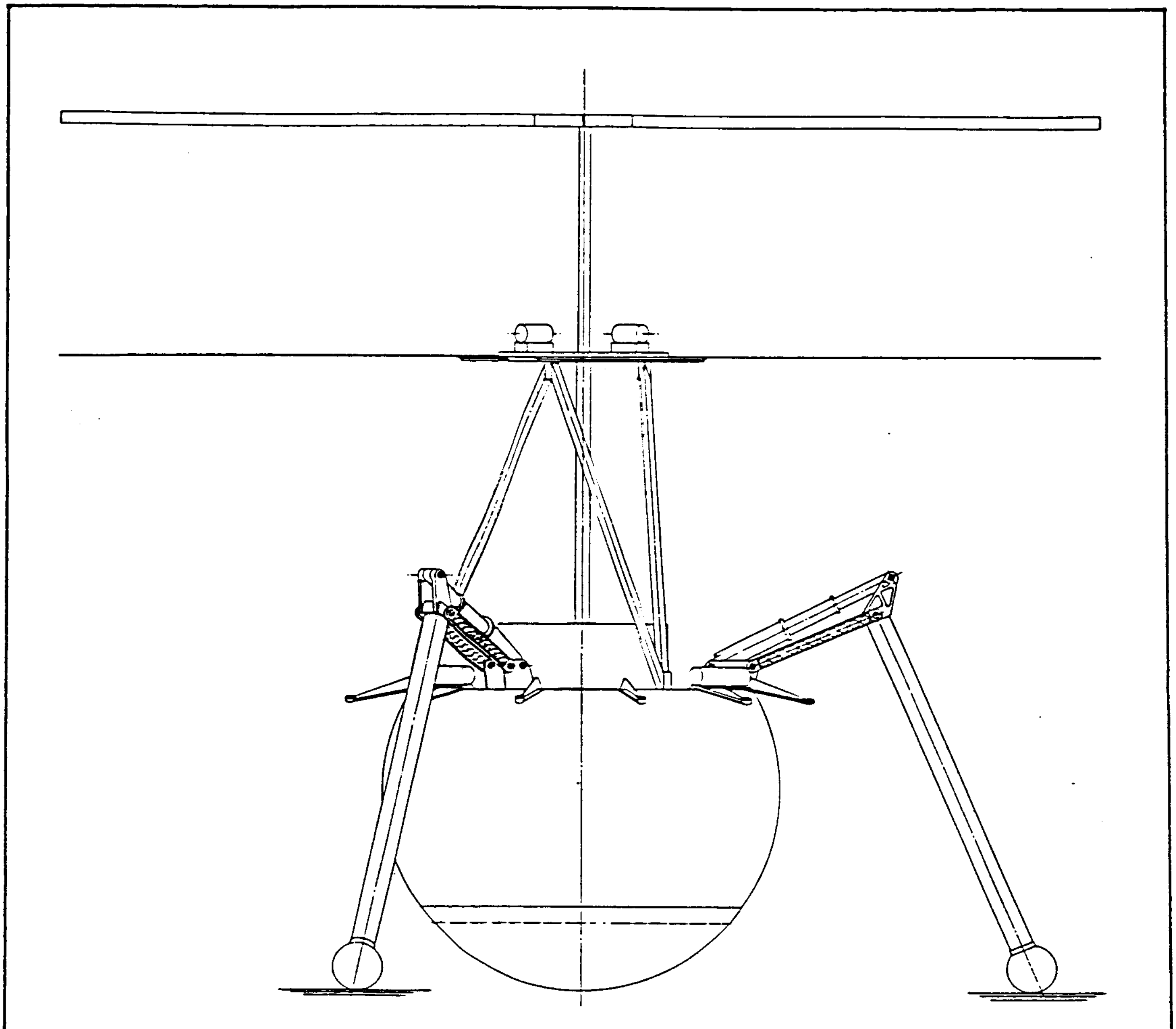

Figure 3-13. Hard Lander Vehicle - Deployed Configuration (JPL Drawing)

This vehicle has a less restrictive pointing requirement at planet contact than the penetrator but it requires a much lower contact speed. As a result, this vehicle has a higher retropropulsion mass for the same landed mass but has lower design requirements (vis-a-vis the terminal acceleration), making subsystems less costly to produce. The overall benefit of this concept lies in the ability to place a small, simple lander on terrain of unknown characteristics with a reasonable expectation of survival.

This concept has only been examined as a feasibility study; no hardware has been built or tested. However, all components are well-understood or have been used elsewhere. The technology readiness of this vehicle concept can thus be rated as moderate.

3.1.4 Penetrator

Previous applications of the penetrator concept have been primarily military in nature. Surveillance devices have been placed in remote or inaccessible locations by means of airdrops or artillery launch. The present suggested application for this concept is to use it as an exploration device for solid surface bodies. Such an application has been studied extensively for use at Mars (Ref. 7).

The basic configuration for a penetrator of this type consists of a high fineness ratio cylinder with a blunted ogive nosecone (Figure 3-14). The afterbody of this vehicle is designed to stop at the surface while the forebody penetrates much more deeply into the surface material. The afterbody thus contains all surface-related experiments (i.e., imaging) and engineering subsystems (i.e., communications). The depth of penetration achieved by the forebody provides several advantages over surface landers. These include superior coupling to the subsurface material for seismic measurements as well as a greater possibility of directly measuring bedrock material rather than inhomogeneous surface material.

One potentially serious drawback of this concept is a requirement to impact the surface at an angle of attack of 10 degrees or less. Surface or buried rocks of irregular shape may deflect the penetrator enough to cause its

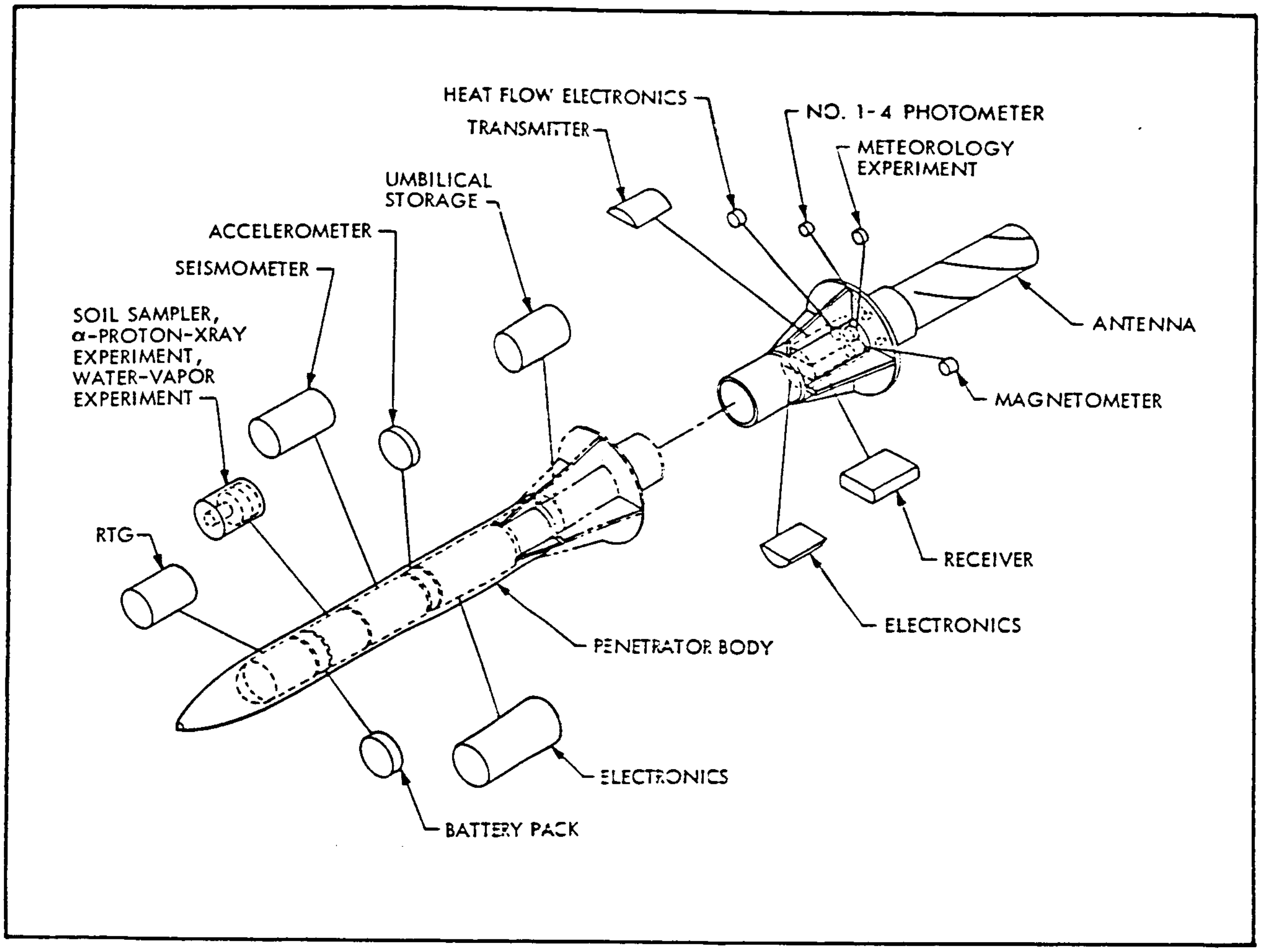

Figure 3-14. Penetrator Vehicle (NASA/Ames Drawing)

destruction although the Mars penetrator study (Ref. 7) indicates that this has a low probability of occurring.

The mass of a system with the characteristics described above is shown in Table 3-9.

Table 3-9

SURFACE PENETRATOR MASS BREAKDOWN

Structure and Devices	29.5 kg
Thermal, Cabling and Pyro	1.0
AACS	(spin-stabilized)
Telecommunications	1.4
Antennas	0.6
Command and Data Handling	1.3
Power Source and Processing	2.3
Launch Tube and Retro Allocation	142.0
Contingency	20.0
Total	198.1 kg

The combination of previously tested hardware and new systems as applied to this concept indicates that the surface penetrator has a moderate to high level of technology readiness.

3.1.5 Chemical Retropropulsion Systems

For the purposes of this study, all-chemical retropropulsion systems are sized to the particular mission performance requirements for which they will be used. As such, these retropropulsion systems are characterized by a set of scaling parameters based on the amount of propellant used. Two different propellant combinations were considered for the candidate missions, each with its own set of scaling parameters. The first of these combinations is an Earth-storable biopropellant system using nitrogen tetroxide (NTO) and hydrazine (N_2H_4). The other combination is a space-storable biopropellant using fluorine (F_2) and hydrazine. Scaling data for these systems, including specific impulse, tankage factor and inert mass, have been developed at JPL (Ref. 27) and will be discussed further in Section 4 of this report. Both systems assume a two-propellant tank arrangement with a 1330 N main engine and a pressure-fed delivery system. The scaling data are valid for propellant loads between 1000 and 5000 kg.

Because of its use on previous missions, the technology readiness of the Earth-storable system is high. The space-storable system, however, has never been used before although it has undergone extensive development and testing. The technology readiness of this second option can thus only be rated as moderate.

3.1.6 Aerocapture

As in the case of the chemical retropropulsion systems, this study assumes that the aerocapture vehicle will be sized based on individual mission requirements. However, all missions will use the same basic aerodynamic configuration. This is the biconic configuration studied extensively at JPL (Ref. 27) as shown in Figure 3-15. A recent study of aerocapture capabilities (Ref. 28) developed a mass estimation algorithm similar in purpose to those used for the chemical retropropulsion systems. The aerocapture scaling algorithm uses the internally-carried spacecraft mass and the entry speed at the target atmosphere as the independent variables. The details of these algorithms will be discussed in Section 4 of this report.

The aerocapture concept clearly requires an atmosphere in order to be feasible, making the outer planets an obvious choice for its use. The aerocapture capabilities study (Ref. 28) indicated that aerocapture is best suited to those missions requiring a large velocity change in order to be captured. Those missions using highly elliptical orbits to tour many smaller satellites may not find any great advantage to the use of aerocapture as opposed to traditional chemical retropropulsion systems. This study also found that large entry speeds such as those which would result from entry at Jupiter and Saturn quickly remove any mass reduction advantage aerocapture may have due to the large amount of ablative thermal protection material required. Thus Titan, Uranus and Neptune appear to be the best candidates among the outer planets for use of this system.

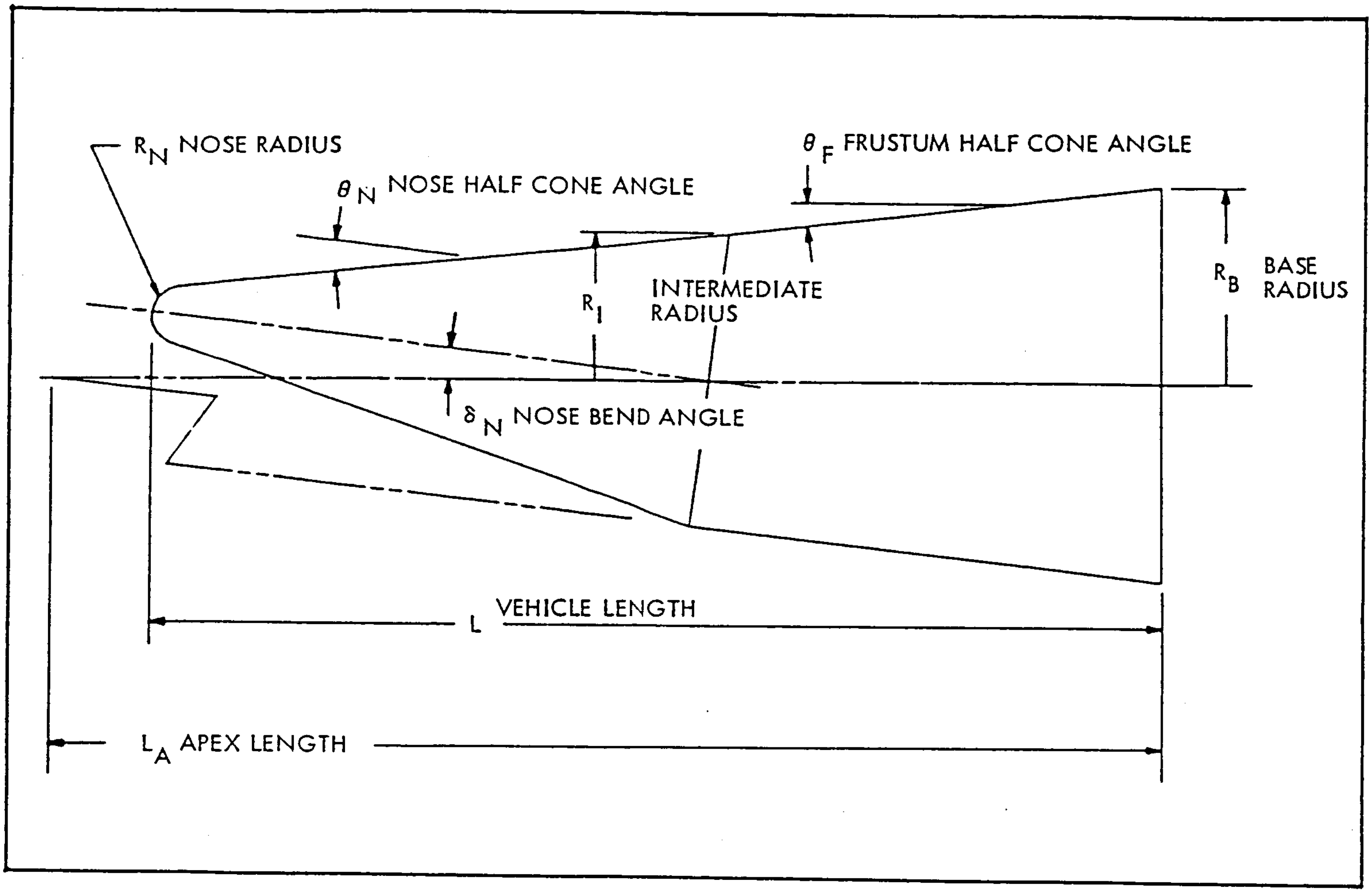

Figure 3-15. Axisymmetric Biconic Aeroshell - Bent Nose Configuration (JPL Drawing)

The biconic concept has been used at Earth for re-entry systems but not on the scale needed for planetary missions. The guidance and control system necessary for the aerodynamic maneuvering conducted by this vehicle is also only in the development stage. Since no actual hardware of the required scale has been built or tested for this concept, the technology readiness must be rated as moderate.

3.1.7 Low-Thrust Propulsion Systems

Two types of low-thrust systems have been investigated as possible upper stages for the candidate missions in this study. Both would use ion bombardment thrusters and mercury propellant, but a different power source would be used for these thrusters. The first type, referred to as nuclear electric propulsion (NEP), uses a nuclear reactor power source which allows operations independent of the distance from the Sun but imposes restrictions on the vehicle in the form of shielding mass and radiation-hardened electronics. The second type uses solar arrays to generate power for the thrusters. This type of propulsion module is generally referred to as a solar electric propulsion (SEP) stage. The operation of this system is limited by the distance from the Sun at which solar arrays can generate sufficient power to drive the thrusters. Thus operating this system much beyond the orbit of Mars is impractical.

A **NEP Stage** consists of two major components. The first is the nuclear reactor power source which is assumed to use an SP-100 (Figure 3-16) system (Ref. 32) as currently being studied by NASA, DOE and DARPA. Since this system is still in the study phase, only general characteristics can be stated regarding its performance. These characteristics are listed in Table 3-10.

Several studies have examined possible designs for the propulsion module which would be supported by the SP-100. These studies have been summarized in Reference 17. The basic design (Figure 3-17) calls for the use of mercury ion thrusters with a specific impulse of 5300 seconds. These thrusters have a lifetime of 1×10^5 amp-hr for a 30 cm diameter version and 2.8×10^5 amp-hr

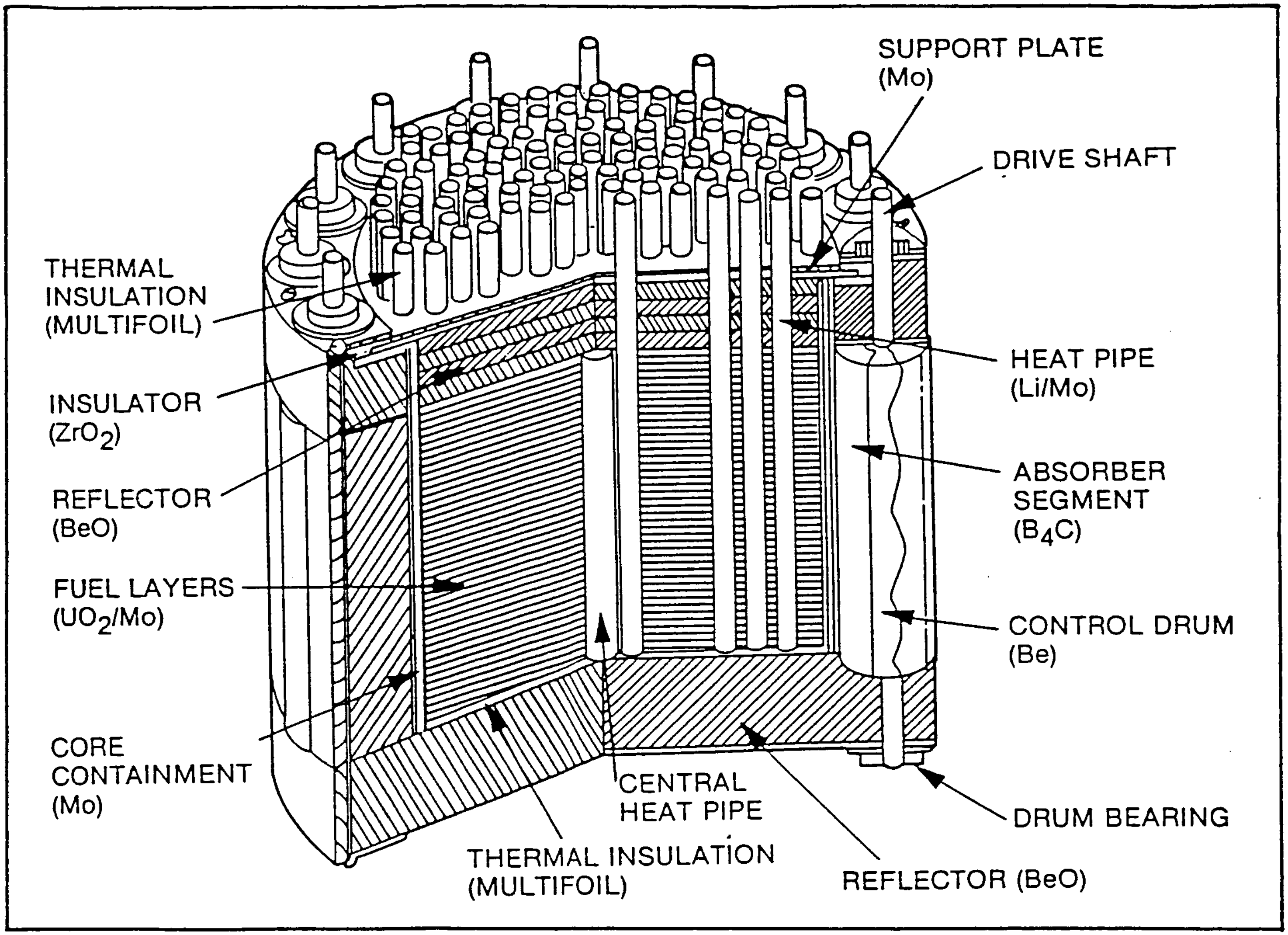

Figure 3-16. Sample NEP Power Source (JPL Drawing)

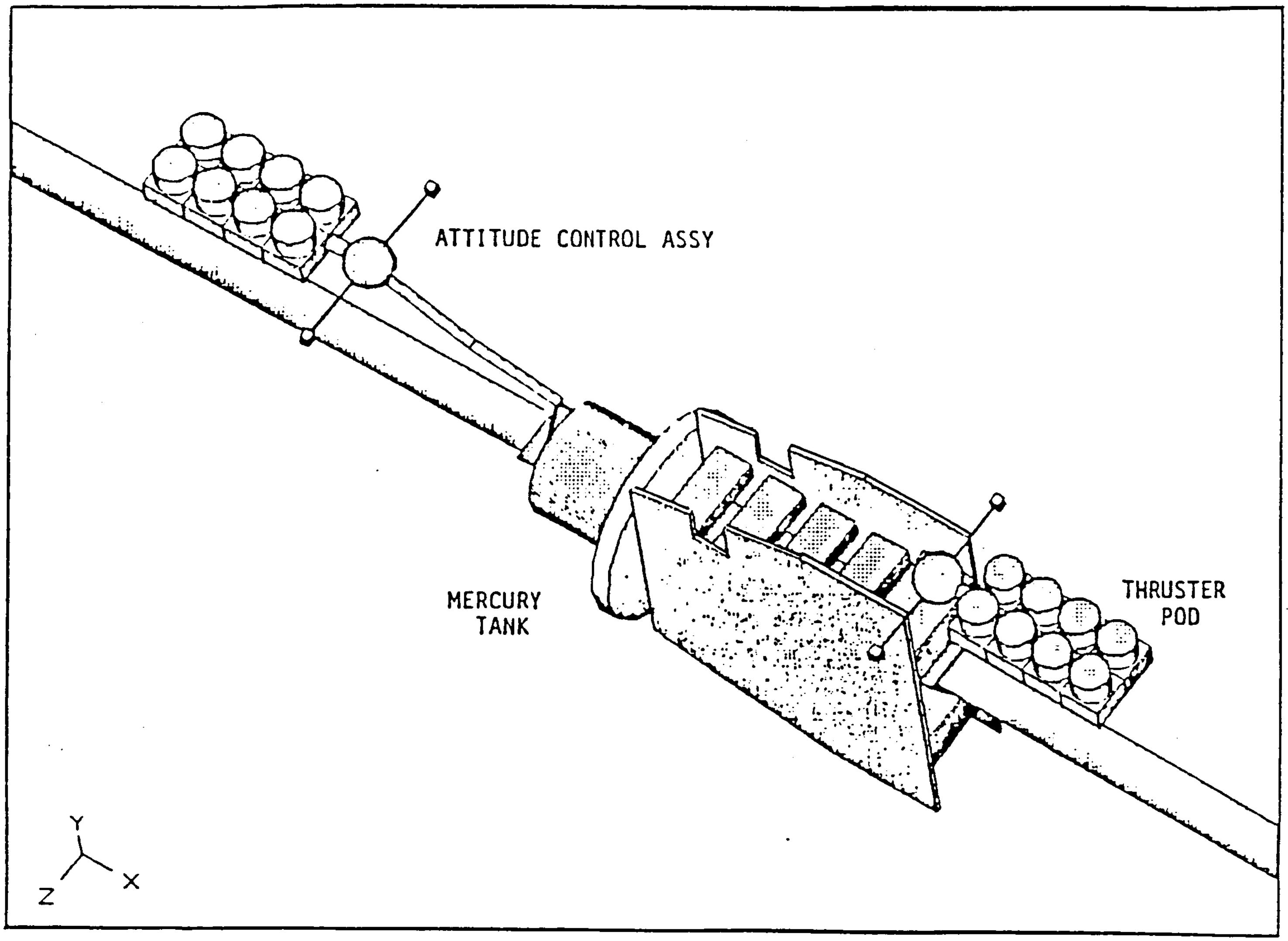

Figure 3-17. NEP Propulsion and Attitude Control Module (JPL Drawing)

Table 3-10

SP-100 NUCLEAR POWER SOURCE CHARACTERISTICS

End of life power - 100 kWe
Mass - 3000 kg
Launch vehicle - NASA Space Shuttle
Launch Configuration - No larger than 1/3 of Shuttle payload bay length
Full power life - 7 years
Total system life - 10 years
Seven year radiation dose at 25 m - 5×10^5 rad (Si); 1×10^{13} neutrons/sq cm

for a 50 cm diameter version. Using projections from other studies, two propulsion modules have been postulated, the mass breakdowns of which are shown in Table 3-11 (Ref. 17).

Thermal control of this module is maintained by single-sided radiators, heat pipes and multilayer insulation. The radiators are sized to dissipate waste heat from the power processors and maintain baseplate temperatures at 50°C. Both systems have approximately 2 kWe available for use by the spacecraft payload.

Development of low-thrust propulsion technology is an ongoing process but an equivalent program for space-based nuclear reactors is just beginning. Thus, the technology readiness of this type of propulsion module is moderate to low.

The **SEP Stage** propulsion system uses the same basic hardware as the NEP stage. However, the power processing for this unit is scaled down to handle the lower power levels generated by the solar arrays. For this study a solar array sized for 32 kWe output at the Earth's radius was assumed. This is a beginning of life (BOL) power level. System losses reduce this to a 28 kWe (BOL) power input to the power processor units and produce a specific impulse

Table 3-11

NEP PROPULSION MODULE MASS BREAKDOWN

Thruster size	**30 cm**	**50 cm**
Thrust per thruster	0.72 N	1.49 N
Maximum number of operating thrusters	4	2
Total number of thrusters	16	6
Maximum number of operating power processors	4	2
Total number of power processors	8	4
Thruster beam current	6.36 Amp	13.1 Amp
Thruster life	15700 hr	21400 hr
Required mission life	51380 hr	51380 hr
Excess subsystem life	22%	25%
Power processor input power	24.3 kWe	49.1 kWe
Thrust Module Masses	**1320 kg**	**738 kg**
Thrusters	192	174
Thruster support structure and actuator	144	132
Power processors	592	176
Power processor radiators and thermal control	360	244
Miscellaneous	32	12
Interface Module Masses	**725 kg**	**503 kg**
Power processing	524	390
Harness	48	18
Thermal control	16	16
Structure	97	59
Thrust subsystem controller	8	8
Miscellaneous	32	12
Propellant tank	**100 kg**	**100 kg**
Power processor radiator area	18 m^2	13 m
Total subsystem mass	**2145 kg**	**1341 kg**
Total required power	97.7 kWe	98.3 kWe

of 3560 seconds from the thrusters. Based on a parametric study for a system with these power levels (Ref. 29), a dry mass for the stage of 1200 kg has been used and Figure 3-18 illustrates one possible configuration (Ref. 33) with a spacecraft payload. The mass of the mercury propellant will be based on the trajectory flown. This type of system has been shown to be capable of

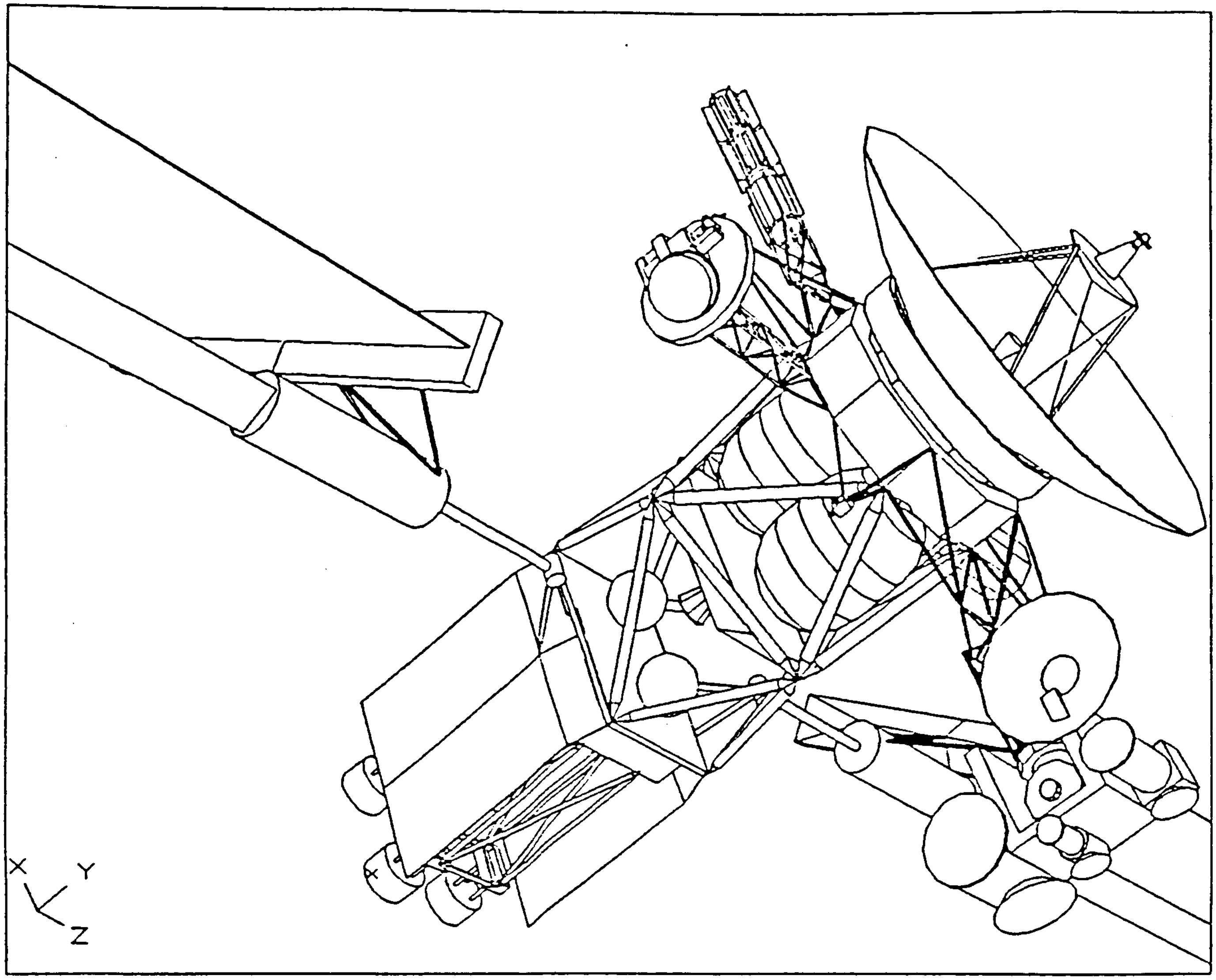

Figure 3-18. SEP Stage with Attached Mariner Mark II Class Spacecraft (JPL Drawing)

maintaining maximum thrust out to a radius of 3.7 AU at which point there is insufficient solar energy to support the stage.

As mentioned for the NEP stage, low-thrust propulsion technology continues to be developed, but for this propulsion module the power source is derived from a well-understood technology. The technology readiness is thus higher than that of the NEP stage since all components have either flown before or are understood; nothing as new as a space-rated nuclear reactor is involved. The technology readiness for this vehicle is moderate to high.

3.2 Assembled Spacecraft Mass Estimates

The various spacecraft elements just discussed will now be used to generate dry spacecraft mass estimates for each of the candidate missions. Only a limited number of options were examined for each mission based on knowledge gained from previous mission studies in this area. For example, low thrust and aerocapture options are known to require longer flight times or a higher injected mass, respectively, to place a spacecraft in orbit around Jupiter when compared to an equivalent ballistic mission. In contrast, ballistic missions to Pluto require significantly longer flight times than a low-thrust spacecraft. As a result only ballistic flights to Jupiter and low-thrust flights to Pluto will be considered. Similar rules of thumb will be applied to the other outer planets as well.

The remainder of this section is made up of a short summary of the vehicle types and mass values which will be required to complete each mission. These values will be used in the next section to compare various flight modes and propellant requirements.

3.2.1 Jupiter Inner Magnetosphere/Polar Orbiter

This mission requires only one vehicle element, a "special" support bus, to carry out the mission objectives. Table 3-12 summarizes the overall dry mass requirements for this vehicle.

Table 3-12

JUPITER INNER MAGNETOSPHERE/POLAR ORBITER MASS REQUIREMENT

Science	69
Bus	431
Total Contingency	50
Injected Mass Requirement (Dry)	**550 kg**

3.2.2 Jupiter Deep Probe/Multiprobe

Three spacecraft elements are needed to complete the science objectives of this mission. These elements consist of a single deep probe capable of reaching 1000 bars, three shallow (Galileo-type) probes and a "special" probe carrier bus. The overall mass estimate for this mission is summarized in Table 3-13, given these vehicle elements.

Table 3-13

JUPITER DEEP PROBE/MULTIPROBE MASS REQUIREMENT

Deep Probe		
Science	28	
Probe Module	197	
Entry Shell	371	
Total Contingency	60	
Subtotal		656
Multiprobe		
Science	55	
Probe Module	87	
Entry Shell	219	
Total Contingency	--	
Subtotal (3 Vehicles)		1083
Support Bus		635
Injected Mass Requirement (Dry)		**2374 kg**

3.2.3 Galilean Satellite Penetrator Network

Two vehicle elements will be required for this mission: three surface penetrators and an orbiter/support bus. The orbiter will be of the Mariner Mark II type. The mission mass requirement is summarized in Table 3-14.

Table 3-14

GALILEAN SATELLITE PENETRATOR NETWORK MASS REQUIREMENT

Penetrator		
Science	1.9	
Penetrator Vehicle and Launch Tube	178.1	
Total Contingency	20.0	
Subtotal (3 Vehicles)		600
Orbiter/Support Bus		
Science	85	
Bus	624	
Total Contingency	71	
Subtotal		780
Injected Mass Requirement (Dry)		**1380 kg**

3.2.4 Europa Orbiter/Lander

This mission is similar to the Galilean Satellite Penetrator Network in that two vehicle elements with comparable science objectives will be required. These two elements are a Mariner Mark II-type orbiter/support bus and a lander, an overall mass summary of which is presented in Table 3-15.

Table 3-15

EUROPA ORBITER/LANDER MASS REQUIREMENT

Lander		
Science	57	
Bus	442	
Total Contingency	57	
Subtotal		556
Orbiter/Support Bus		
Science	85	
Bus	687	
Total Contingency	77	
Subtotal		849
Injected Mass Requirement (Dry)		**1405 kg**

3.2.5 Titan Orbiter/Penetrator Network

As with the Galilean Satellite Penetrator Network, this mission will require a single Mariner Mark II-type orbiter and several penetrators to complete the science objectives. A previous study (Ref. 27) has shown that aerocapture is the best means of delivery to this planet and this capture mode will be assumed here. The mass estimate for the aerocapture vehicle will depend on the trajectory flown and thus will be provided in a later section. An attempt was made to optimize the vehicles for this particular mission and delivery mode which accounts for the slight difference in mass of the mission elements. Table 3-16 summarizes the mass estimates for these elements.

Table 3-16

TITAN ORBITER/PENETRATOR NETWORK MASS REQUIREMENT

Penetrator		
Science	2.1	
Penetrator Vehicle and Launch Tube	87.9	
Total Contingency	10.0	
Subtotal (3 Vehicles)		300
Orbiter		
Science	131	
Bus	548	
Total Contingency	60	
Subtotal		739
Orbited Mass Requirement (Dry)		**1039 kg**

3.2.6 Titan Orbiter/Buoyant Station

In terms of the number of different vehicle elements needed, this is the most complex mission in the candidate set. Six different vehicle types will be required to carry out this mission. In addition, two separate launches will be used to place all elements in position. An aerocapture vehicle will be used to deliver an orbiter of the Mariner Mark II type and several small haze probes. Again the aerocapture vehicle mass will depend on the trajectory flown and thus will be determined later. A special probe carrier on a flyby trajectory will be used to deliver four entry probes of two different types.

The masses for all of these elements except the aerocapture vehicle are summarized in Table 3-17.

Table 3-17

TITAN ORBITER/BUOYANT STATION MASS REQUIREMENT

First Launch		Second Launch	
Orbiter		**Buoyant Station**	
Science............. 131		Science.............. 80	
Bus................. 548		Probe Module......... 334	
Total Contingency... 60		Entry Shell and Gas Transport...... 507	
		Probe Launch System.. 50	
Subtotal...............	739	Subtotal.................	971
Haze Probes (3) and Launch System.........	295		
		Balloon Probes	
Orbited Mass Requirement (Dry)......	**1034 kg**	Science.............. 20	
		Probe Module......... 107	
		Entry Shell and Gas Transport...... 169	
		Probe Launch System.. 10	
		Subtotal (3 Vehicles) ...	918
		Probe Carrier (with Contingency).............	454
		Injected Mass Requirement (Dry)......	**2343 kg**

3.2.7 Saturn Ring Rover

The demanding nature of the trajectory to be flown during this mission indicates that only a NEP stage can be used. Thus two vehicle elements will be needed to complete the mission objectives: a NEP stage and a NEP spacecraft bus. Table 3-18 summarizes the dry mass requirement for this vehicle.

Table 3-18

SATURN RING ROVER MASS REQUIREMENT

Science	141
Bus	1184
Total Contingency	175
Nep Stage (Dry)	2145
Injected Mass Requirement (Dry)	**3645 kg**

3.2.8 Uranus Orbiter/Probe

Two different vehicle elements are needed to carry out this uprated Core Program mission. These include a Mariner Mark II orbiter/support bus and an atmospheric entry probe which can survive to the 30 bar pressure level. This vehicle could be delivered into orbit by means of an aerocapture vehicle or a chemical retropropulsion system. These options will be discussed further in the following sections. Table 3-19 summarizes the mass of those elements which must be transported to Uranus regardless of the option chosen.

Table 3-19

URANUS ORBITER/PROBE MASS REQUIREMENT

Atmospheric Probe		240
Orbiter		
Science	60	
Bus	756	
Subtotal		816
Delivered Mass Requirement (Dry)		**1056 kg**

3.2.9 Uranus Orbiter/Multiprobe

This mission requires three vehicle elements to complete the science objectives. These elements include a Mariner Mark II-type orbiter/support bus, three entry probes capable of withstanding 100 bars of pressure, and an aerocapture vehicle. With the exception of the aerocapture vehicle, which will be discussed later, the major vehicle elements for this mission are summarized in Table 3-20.

Table 3-20

URANUS ORBITER/MULTIPROBE MASS REQUIREMENT

Multiprobe		
Science	54.5	
Probe Module	186.6	
Entry Shell	106.1	
Subtotal (3 vehicles)		1042
Orbiter		
Science	93	
Bus	669	
Contingency	76	
Subtotal		838
Orbited Mass Requirement (Dry)		**1880 kg**

3.2.10 Neptune Orbiter/Dual Probe

Four vehicle elements are required by this mission to complete the science objectives. One of these elements is either a NEP stage or an aerocapture vehicle depending upon the flight mode selected. The three remaining elements are the support bus (either a Mariner Mark II type or a NEP type), a probe targeted for Triton (20 bar pressure capability) and a probe for Neptune (100 bar pressure capability). The mass values for each of these elements are summarized in Table 3-21.

Table 3-21

NEPTUNE ORBITER/DUAL PROBE MASS REQUIREMENT

	Aerocapture		NEP	
Triton Probe				
Science	53.6		53.6	
Probe Module	101.2		101.2	
Entry Shell	64.9		64.9	
Adapter	11.3		11.3	
Subtotal		231		231
Neptune Probe				
Science	30.7		30.7	
Probe Module	194.3		194.3	
Entry Shell	106.0		106.0	
Adapter	28.0		28.0	
Subtotal		359		359
Orbiter				
Science	93.0		93.0	
Bus	669.0		1184.0	
Contingency	76.0		175.0	
Subtotal		838		1452
Orbited Mass Requirement (Dry)		**1428 kg**		**2042 kg**

3.2.11 Pluto Orbiter/Lander and Charon Lander

The final mission to be described from the candidate set is a reconnaissance mission to investigate Pluto and Charon. This mission will require three vehicle elements to meet all science objectives: a NEP stage, a NEP spacecraft bus, and two hard landers. Table 3-22 summarizes the mass estimate for this configuration.

Table 3-22

PLUTO ORBITER/LANDER AND CHARON LANDER MASS REQUIREMENT

Hard Lander		
Science	5.0	
Bus	28.6	
Total Contingency	3.0	
Subtotal (2 Vehicles, Dry)		73
NEP Spacecraft		
Science	105	
Bus	1184	
Total Contingency	180	
Subtotal		1469
NEP Stage (Dry)		2145
Injected Mass Requirement		**3687 kg**

4. PAYLOAD DELIVERY PERFORMANCE TRADES

4.1 Introduction

This section focuses on the trajectory and mass delivery aspects of the proposed outer planet advanced mission concepts. Mission performance capability is a function of many parameters, including: launch vehicle/upper stage, interplanetary flight mode, trajectory type (which may be a function of launch year), the method of orbit capture, and the type of retropropulsion used for post-launch mission phases. In general, the performance stated in this section is shown as net payload into orbit versus flight time for the various parameters listed above. Thus, given the mass requirements of the particular mission, as shown in the previous section, a minimum flight time for each candidate mission for each relevant set of parameters is obtained.

4.1.1 Chemical Upper Stage Definition

The first step in calculating mission performance is to determine the mass which can be injected into an interplanetary trajectory. This requires a definition of upper stage choices to be examined, and calculation of injected mass versus launch energy for the selected set. For this study, a group of launch vehicles based upon the family of Centaur upper stages was selected. A derivative of a recent Marshall Space Flight Center (MSFC) Orbital Transfer Vehicle (OTV) concept was added to this group. The relevant parameters are shown in Table 4-1.

The first two Centaur stages considered are for use with the 65K Shuttle on a standard launch to a 130 n.mi. parking orbit. The less capable of the two, the (short) Air Force version of the wide tank Centaur, is designated (G). The Centaur(G') is the (long) NASA version of the same vehicle. In the case of the Centaur(G'), propellant off-loading is required due to the 65,000 lb_m cargo mass limitation of the Shuttle and therefore, the performance will be degraded, particularly at the lower C_3 ranges. A third option, then, is a fully-loaded Centaur(G'), designated OOA for on-orbit assembly. No attempt is made here to ascertain whether this vehicle is fueled at a Space Station refueling depot and launched from the Space Station, or whether the tanks are simply topped off in orbit by a Shuttle astronaut servicing crew. In either

Table 4-1

CHEMICAL UPPER STAGE PARAMETERS

UPPER STAGE	THRUST LEVEL (N_f)	SPECIFIC IMPULSE (sec)	MAX PROPELLANT (kg_m)	BURNOUT[1] WEIGHT (kg_m)	ASE[2] (kg_m)
STAR 48	67604	290.9	2025	289	--
CENTAUR(G)	146791	440.5	13255	2719 (2925)	3705
CENTAUR(G')	146791	446.4	20197	2936 (3134)	4253
OTV(4-R)/	66723	482.5	25609[3]	2894	--
OTV(2-E)	66723	482.5	12882	1297	--

(1) Parenthetical value applies when used as 1st stage

(2) ASE applies only to Shuttle (65K) launches

(3) OTV(4-R) launched to $\leq$ 24 hours orbit, 155 kg_m propellant reserved for return ΔV.

case, the performance advantage of the fully-loaded tanks is acquired. (Proper injection timing is assumed, i.e., no plane change penalty for a Station launch is assumed here.) For all three launch vehicles, the performance advantage at the higher launch energies of a kick stage is shown and also used in the performance calculations. This insertion module, designated IM, is based upon the Star 48 solid rocket motor.

The fourth launch vehicle considered is a two-stage vehicle comprised of mating fully loaded Centaur(G') and Centaur(G) upper stages. This vehicle is designated the OOA Centaur(G')/Centaur(G).

The final and most capable launch vehicle considered is a conceptual propulsion system based upon recent space-based OTV studies at MSFC. The first stage is a 4-tank reusable OTV (designated 4-R) which is launched into a maximum 24-hour orbit. The expended stage then returns to a Space Station compatible orbit by means of aeromaneuvering in the Earth's atmosphere, followed by a propulsive perigee raise manuever. The second stage is a 2-tank expendable version of the same vehicle, designated 2-E, and is launched to escape with the spacecraft from perigee of the first-stage orbit.

Injected mass performance of these five launch vehicles is shown in Figure 4-1. These data were generated using the parameters from Table 4-1 in SAIC's STAGE program which accounts for finite-thrust gravity losses.

4.1.2 Orbit Capture Modes

For this study, two methods of effecting orbit capture were employed: (1) chemical stages for performing the larger retropropulsion maneuvers, and (2) aerocapture technology.

Chemical Retropropulsion Stage Definition. Chemical retropropulsion systems using two types of propellants were considered in this study. The Earth-storable and space-storable systems are based upon recent concept studies at JPL and their relevant parameters are summarized in Table 4-2. Both are pressure-fed systems.

Table 4-2

CHEMICAL RETROPROPULSION STAGE PARAMETERS

TYPE	PROPELLANT	I_{sp} (sec)	TANKAGE FACTOR	INERT MASS (kg)
E-S	NTO/N_2H_4	315.0	0.1332	69.9
S-S	F_2/N_2H_4	370.0	0.135	154.6

A reaction control system (RCS), operating at a lower thrust level for impulses lower than 0.150 km/sec is included in these stage definitions. The RCS operates at an I_{sp} of 50 seconds below the nominal for these maneuvers, midcourse navigation and orbiter stationkeeping.

Aerocapture. Most recent studies of aerocapture have assumed the use of a biconic vehicle with a moderate L/D. This vehicle makes a single deep pass through the atmosphere at the periapse of its approach trajectory. Sufficient

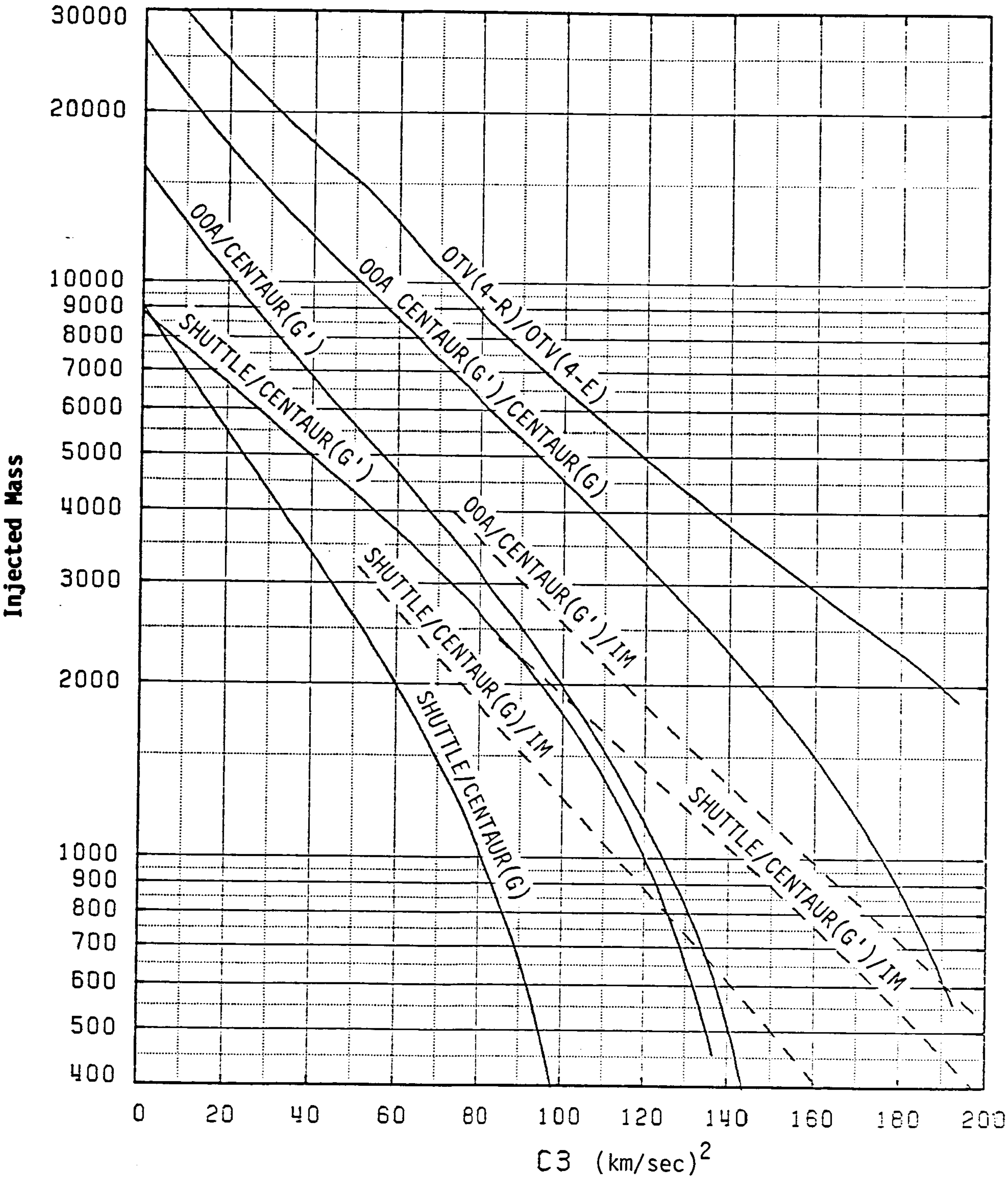

Figure 4-1. Launch Vehicle Performance

kinetic energy is removed by aerodynamic drag to capture the vehicle and place it on a transfer orbit to its final operational altitude. A report (Ref. 28), which reviews previous design studies for interplanetary aeromaneuvering vehicles, has been prepared to assist mission planners in making the design trade-offs.

The scaling laws developed (Ref. 28) for determining aeroshell mass and subsequent performance are shown in Table 4-3.

Table 4-3

AEROCAPTURE VEHICLE SCALING LAWS

1. Determine the spacecraft mass in kilograms.

2. Determine the entry velocity in km/sec; if the velocity is less than 5.86 km/sec, set V_e to 5.86 km/sec.

3. Calculate the structural mass from

$$M_{str} = M_{S/C} [0.003536 V_e + 0.04704].$$

4. Calculate the thermal protection system mass from

$$M_{tps} = M_{S/C} [0.2006 (V_e - 5.86)^{0.3451} + 0.06233].$$

5. Calculate the flap mass from

$$M_{flap} = 1.638 M_{S/C} [0.008913 + 0.006431 \ln (V_e)].$$

6. Find the total aeroshell mass from

$$M_{a/s} = M_{str} + M_{tps} + M_{flap} + M_{aux\ syst}.$$

The auxiliary systems mass has a nominal budget of 100 kg and includes such items as navigation sensors, thrusters, flap actuators, external communication antennas and attitude-control propellant.

4.1.3 Interplanetary Flight Modes/Trajectory Types

Three flight modes are considered for missions analyzed in this study: ballistic and two modes utilizing low-thrust propulsion, SEP and NEP. The delivery options are further increased by examining both direct and indirect trajectory types, as well as Jupiter gravity-assist swingbys. The NEP flight mode is the only delivery option considered for the Pluto orbiter mission.

Jupiter Gravity-Assist Swingbys. For missions to the outer planets, trajectories utilizing Jupiter gravity-assist swingbys can greatly reduce launch energy requirements and mission trip times. Jupiter swingbys to Uranus and Neptune occur at 14 and 13 year intervals, respectively, with opportunities for each occurring in 1992-1995 and again in the 2004-2007 time frame. The opportunity for a Jupiter swingby to Saturn occurs at 20 year intervals. A J/S launch opportunity does occur in the late 1990's and is followed by the next opportunity in approximately 2015, the earliest opportunity in the 21st century. Unfortunately, this latter date is probably too late for the time frame of the missions proposed in this study.

Direct Flight. Direct ballistic missions are employed for both Jupiter and Saturn missions. However, direct ballistic missions to Uranus and Neptune are not considered due to the excessive launch energy requirements and lengthy flight times. Instead, advantage was taken of Jupiter swingby opportunities to Uranus and Neptune making the J/U and J/N delivery options the baseline ballistic trajectories for these missions.

Indirect Flight. The only SEP trajectory type included here is the 2^+ SEEGA, and it is examined for missions to Uranus, Neptune and Pluto. In addition, ΔVEGA delivery options are considered for missions to all targets, except Pluto (as mentioned previously). The advantage of the indirect trajectory types is the ability to capture the mission with a less capable launch vehicle due to the decreased launch energy requirements. However, these trajectory types will, in general, increase the total trip time by the time spent on the Earth-to-Earth leg. Two and three-plus (2^+ and 3^+) ΔVEGA's are used for both Jupiter and Saturn missions respectively. For Uranus and

Neptune, J/U and J/N ΔVEGA missions are employed to complement the J/U and J/N ballistic missions. Also, J/U and J/N SEEGA trajectories are examined for these far outer planets.

The SEP stage used in the SEEGA trajectory calculations assumes 32 kw of array power (BOL) with 28 kw of power (BOL) input to the Power Conditioning Units. This stage assumes an I_{sp} of 3560 seconds and a total system efficiency of 68.2%. The total dry mass is 1200 kg, and the maximum thrusting distance is approximately 3.7 AU.

NEP Flight. The great potential of NEP application to outer planet missions lies in the fact that the nuclear reactor power source operates independently of the distance from the sun. This characteristic of useful thrust acceleration at a large distance allows the vehicle system to slow down near the target planet ($V_{HP} \approx 0$) and achieve orbit capture with relatively small propellant expenditure.

Another feature of NEP is that it may be employed during planetocentric operations; i.e., to spiral out from Earth orbit to escape conditions and to spiral in to planet capture orbit without any intervening phase of chemical propulsion. The nominal Earth-escape spiral begins from a 700 km orbit (nuclear-safe altitude) and terminates when $C_3 = 0$ energy conditions are attained. The 65K Shuttle can deliver a maximum of 20,000 kg to a 700 km orbit, using two orbital maneuvering system (OMS) kits, and at this maximum initial mass, the spiral escape time is 535 days. Target planet spiral orbit capture is assumed for all NEP missions in this study, and Earth-escape spirals are considered in addition to launching the NEP stage and spacecraft to $C_3 > 0$.

The NEP stage used in this study was adopted from recent SP-100 work conducted at JPL (Ref. 17). The reactor generates 100 kw of electrical power, and the thrusters operate at an I_{sp} of 5500 sec and efficiency of 0.776. The dry stage mass is 5145 kg, which includes thruster/power conditioning redundancy to achieve the required operating lifetime. NEP trajectory data were generated using the CHEBYTOP computer program.

4.1.4 Probe Deployment

The following criteria for probe deployment are based upon detailed performance trades conducted in a previous study (Ref. 17).

NEP. Probes are deployed after orbit insertion; therefore, the required payload into orbit includes the orbiter plus probe(s) mass.

Aerocapture. The use of aerocapture for orbit insertion mandates probe deployment from orbit. As in the case of NEP, the required payload into orbit includes the orbiter mass plus the sum of all probe masses.

For the remaining flight modes, trajectory types, and orbit capture methods the probe(s) were deployed on approach to the target. Hence the required payload into orbit for these cases is the orbiter mass only. In all cases, a ΔV budget of 50 m/sec for bus deflection following probe deployment is allocated in addition to 100 m/sec for orbiter stationkeeping following orbit insertion and all probe deployments.

4.2 Jupiter

The mass performance data for the four proposed Jupiter missions are shown in Figures 4-2 through 4-6. In all cases, performance is displayed for both direct and ΔVEGA ballistic trajectory types, launched in 2000 and 1998 respectively, and for the relevant subset of launch vehicle/upper stages. A characteristic common to the performance data for all the missions is that the ΔVEGA trajectory type adds approximately two years to the mission time of a direct ballistic mission for the same upper stage, but it does enable the mission to be captured using a less capable upper stage.

The Inner Magnetosphere/Polar Orbiter mission performance is shown for both a single and dual orbiter concept in Figures 4-2 and 4-3, respectively. The performance for each mission is also shown for two extreme orbit sizes in order to bound the performance. The single orbiter performance is shown for 1.014 x 6 R_J and 1.014 x 20 R_J orbits. Note that the Shuttle/Centaur(G') can not capture the mission into the tighter orbit, even utilizing the ΔVEGA. The

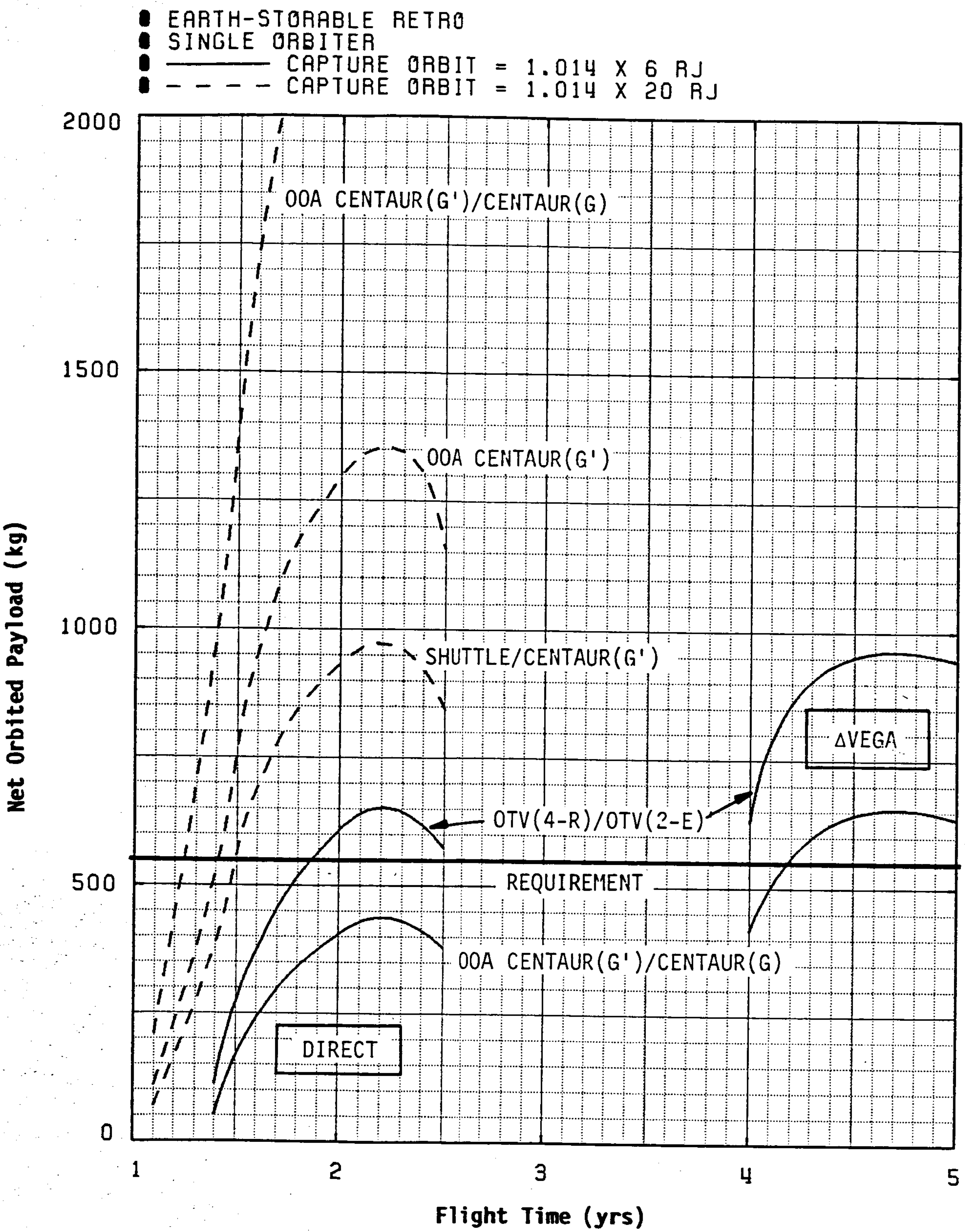

Figure 4-2. Jupiter Inner Magnetosphere/Polar Orbiter Mission Performance

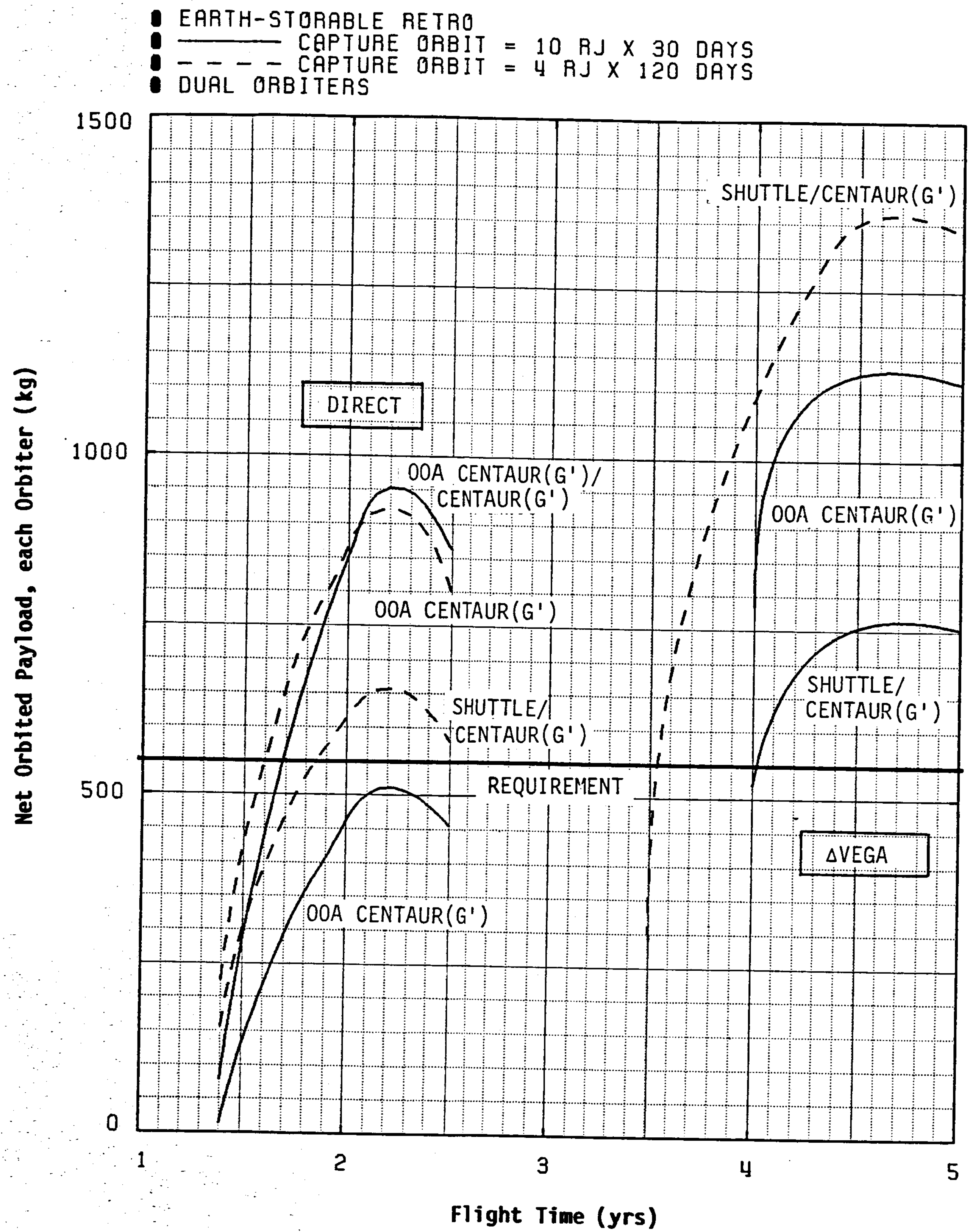

Figure 4-3. Jupiter Inner Magnetosphere/Polar Orbiter Mission Performance

dual orbiter performance is shown for 10 R_J x 30 day and 4 R_J x 120 day orbits, and the performance shown is the net orbited payload mass of each orbiter. For this mission the Shuttle/Centaur(G') can capture the mission into the tighter orbit by using the ΔVEGA option.

The Jupiter Deep Probe/Multiprobe mission performance is shown in Figure 4-4. The 656 kg deep probe is released prior to orbit insertion but the three smaller 333 kg probes are released from the 5.3 R_J x 90 day orbit, hence the required payload into orbit of 1635 kg. An Io gravity-assist is employed during the orbit insertion maneuver to reduce the impulse required. At a minimum, the stacked Centaur is required to capture this mission on a direct ballistic trajectory, while a ΔVEGA option, using a Shuttle/Centaur(G'), enables the mission at a flight time of slightly over four years.

The next two missions terminate with a Europa orbiter. The first is a Galilean Satellite Penetrator Network which utilizes a Ganymede gravity-assist into Jupiter orbit, followed by a low-V_∞ satellite tour, deploying 200 kg penetrators along the way at Callisto, Ganymede and Europa, ending with insertion into a 500 km circular orbit about Europa. The performance for this mission is shown in Figure 4-5, and requires a fully-loaded Centaur(G'), even in the ΔVEGA mode.

The last mission is the Europa Orbiter/Lander mission. The performance for this mission is shown in Figure 4-6. This figure illustrates the fact that this is the most demanding of all the Jupiter missions in terms of mass required into a 500 km circular orbit about Europa. As before, a Ganymede gravity-assisted Jupiter orbit insertion maneuver is employed, followed by the special low V_∞ tour. The 556 kg lander, plus its retropropulsion, is deployed from Europa orbit leading to the requirement of 1945 kg in Europa orbit.

The details of the Galilean satellite low V_∞ tour utilized in these last two Jupiter missions are discussed fully in Reference 6.

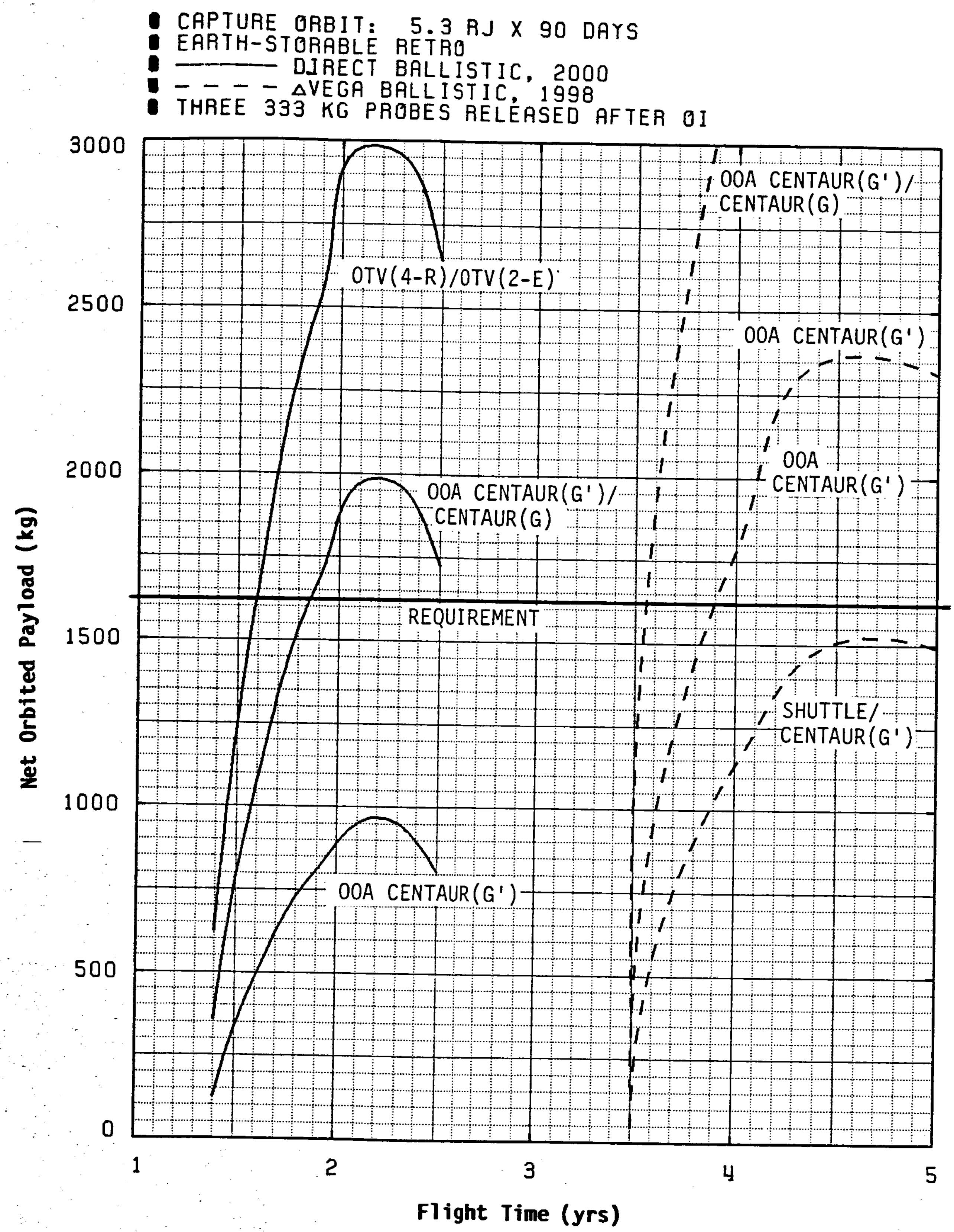

Figure 4-4. Jupiter Deep Probe/Multiprobe Mission Performance

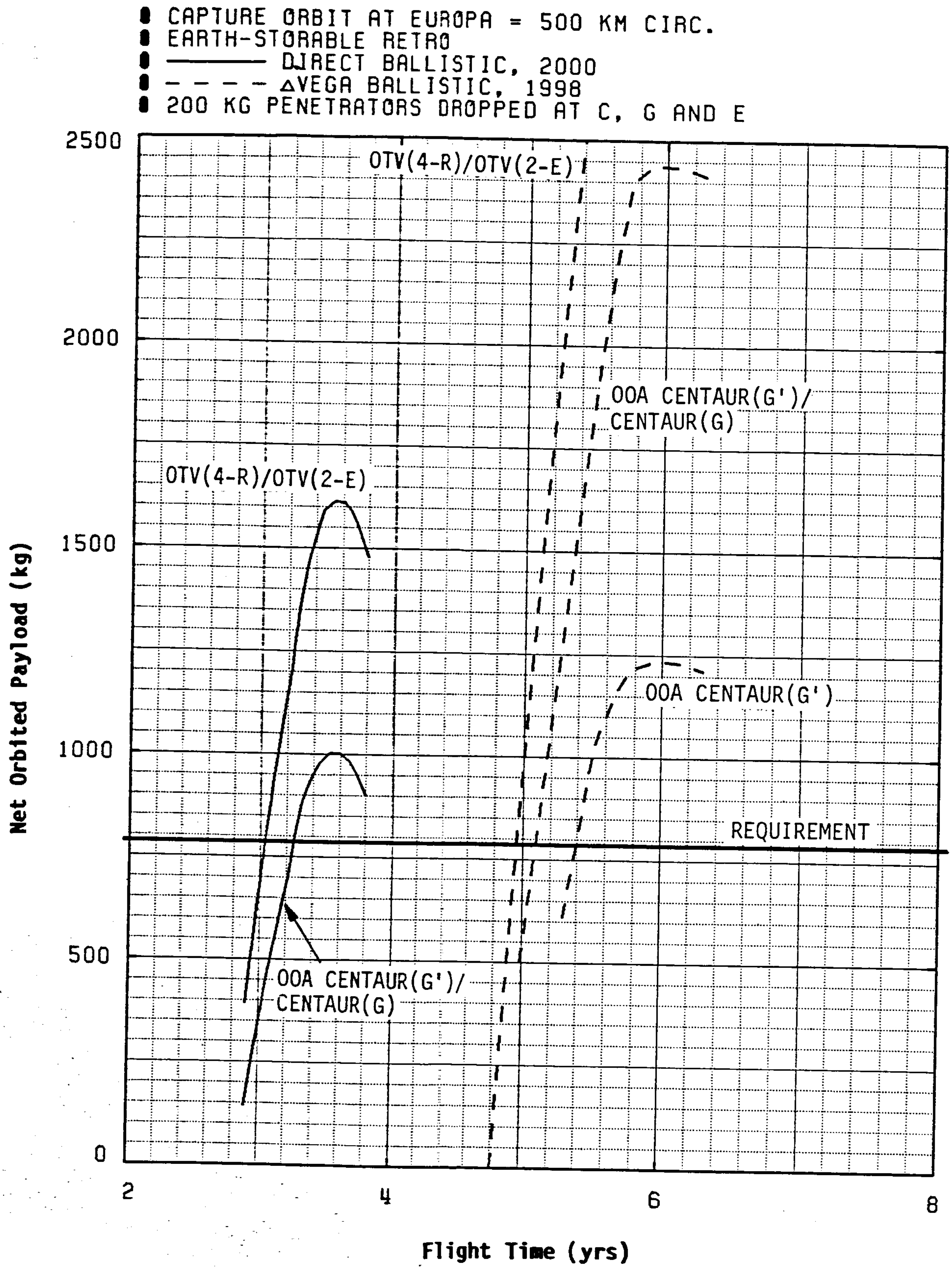

Figure 4-5. Galilean Satellite Penetrator Network Mission Performance

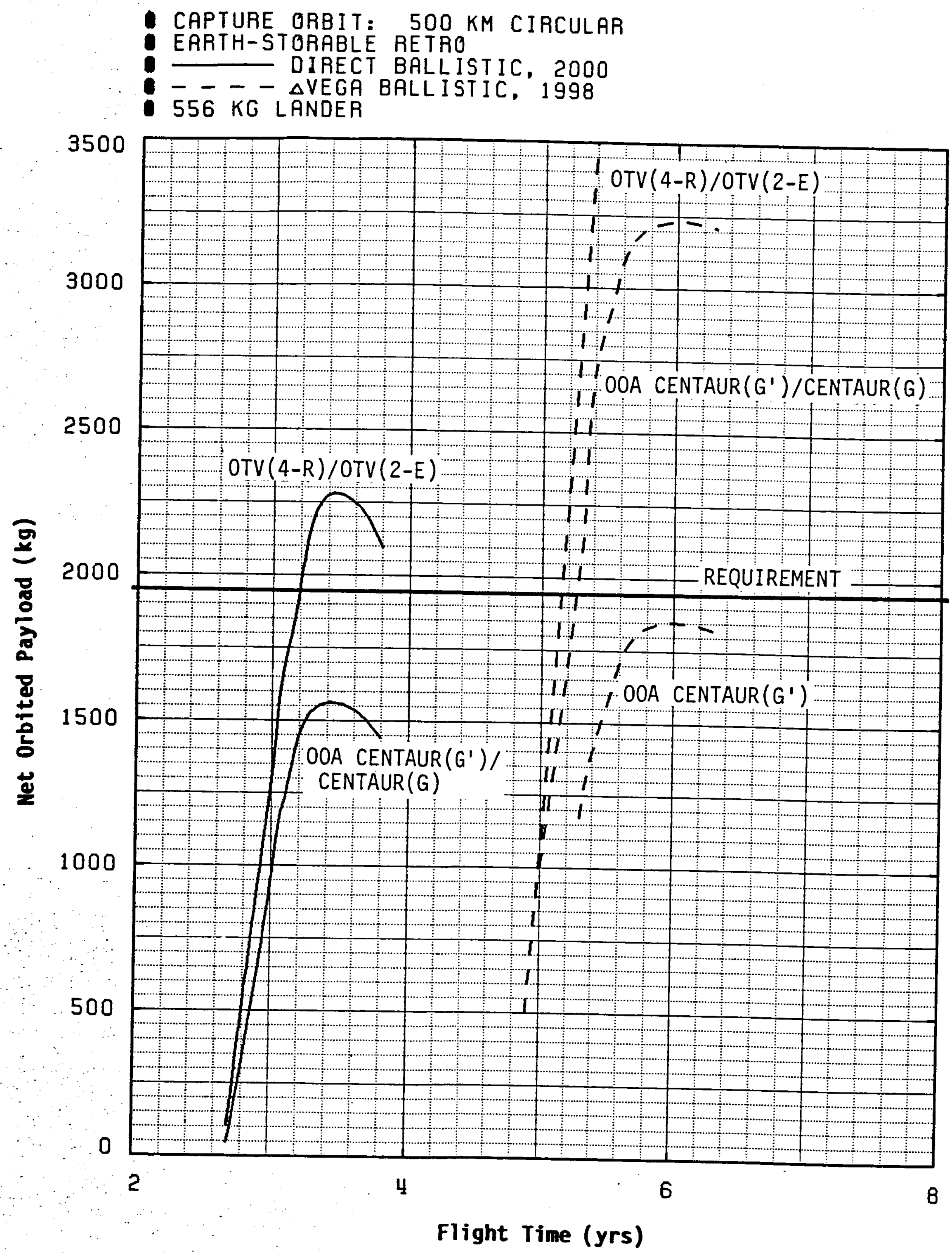

Figure 4-6. Europa Orbiter/Lander Mission Performance

4.3 Saturn

The Saturn missions proposed in this study fall into two categories: (1) Titan-intensive missions; and (2) the Saturn Ring Rover mission. A detailed study of future Titan exploration using advanced concepts has previously been completed (Ref. 21), and the detailed mission analyses are presented there. Figure 4-7 shows mission performance capability as net payload into a 1000 km circular orbit about Titan for the two Titan orbiter missions studied here. The orbiter bus in each mission is carrying penetrator/probes weighing 300 kg and the performance was calculated accordingly; hence, the different required mass into orbit for the two capture modes, Earth-storable and aerocapture. Aerocapture at Titan is necessary to capture the mission with a direct or J/S trajectory type using the OOA Centaur(G') upper stage. A Shuttle/Centaur(G') combination can capture the mission by using a Jupiter swingby trajectory in conjunction with aerocapture at Titan. Recall, however, that the Jupiter swingby opportunities to Saturn occur in the mid-to-late 1990's, and again in the 2010-2015 time frame. The utility of both the J/S and J/S ΔVEGA flight modes is questionable for the timing of the Titan missions considered here, but they are included for purposes of comparison.

A detailed analysis of the Saturn Ring Rover mission concept was conducted at JPL as part of the SP-100 study of nuclear power requirements for future civil missions. The details of four mission concepts are included in Reference 17. Due to the non-Keplerian motion of the spacecraft orbit about Saturn, NEP is required.

An event time summary for the reference mission is shown in Table 4-4. The total initial mass required in the nuclear-safe orbit about Earth for this mission is slightly less than 17000 kg, which provides a Shuttle launch margin of 3000 kg. The mission employs spiral maneuvers on each end of the mission and the total mission duration is 10.4 years.

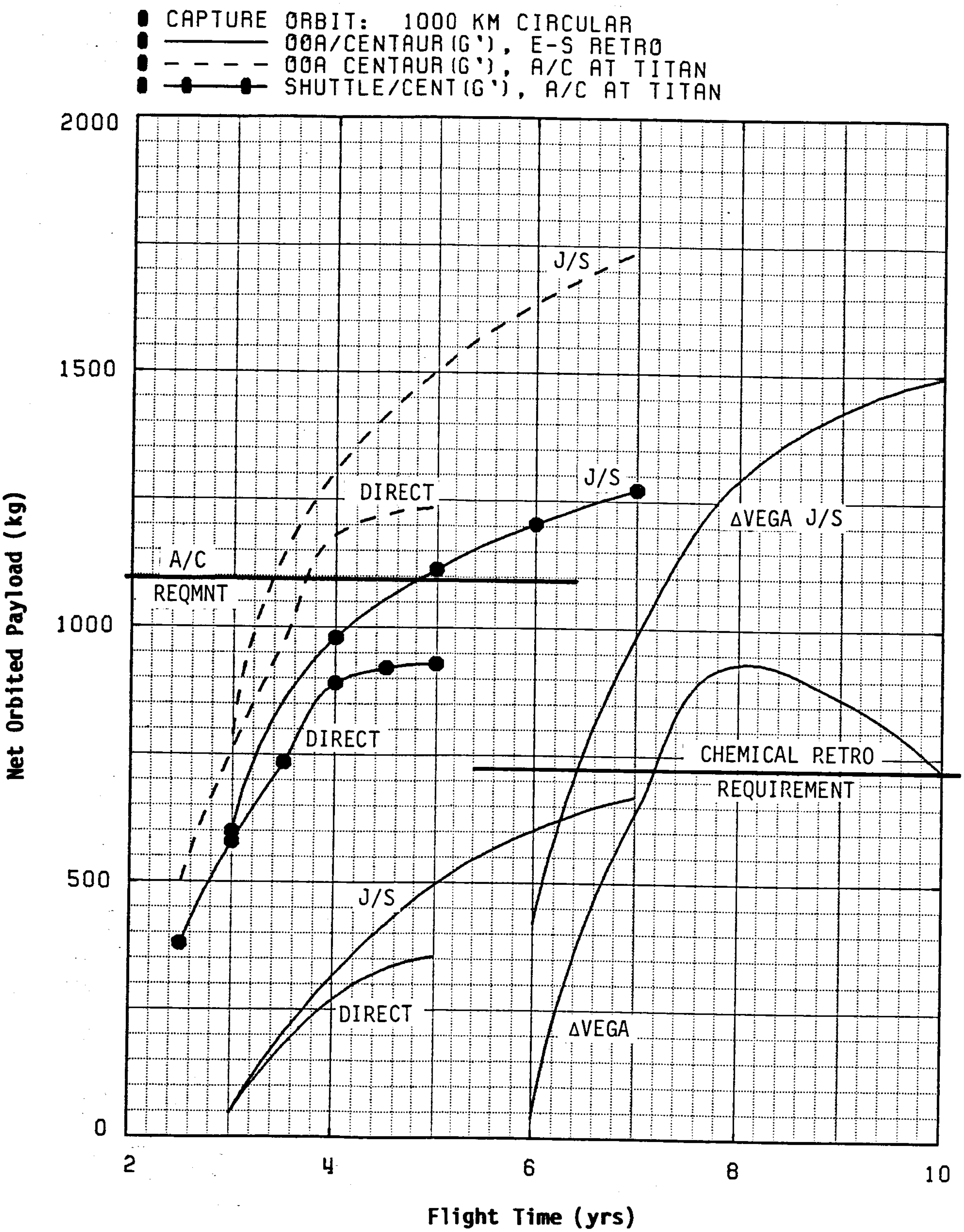

Figure 4-7. Titan Orbiter Mission Ballistic Performance

Table 4-4

EVENT TIME SUMMARY FOR SATURN RING ROVER MISSION

Event	Time (days)
Launch	0
End of escape spiral	425
Start of first coast	660
End of first coast	885
Start of second coast	1392
End of second coast	2842
Saturn arrival and start of capture spiral phase	3075
Titan encounter	3176
Start of ring rendezvous	3502
End of mission	3810
Total mission duration (years)	10.43

Ring rendezvous phase starts at 3.0 R_S

End of ring rendezvous phase and mission is at 1.1 R_S.

(from Ref. 17)

4.4 Uranus

4.4.1 1995 Orbiter/Probe Mission

Due to the near-term nature of this mission, only the ballistic flight mode is assumed as a delivery option for both J/U ballistic and J/U ΔVEGA; only the Centaur family of launch vehicles is considered as a realistic option. The mission performance is summarized in Table 4-5. In all cases, the 240 kg probe is deployed upon approach to Uranus. Note that the total trip time is reduced by approximately two years for each increase in launch vehicle capability for the J/U ballistic mission. The effect is less for the J/U ΔVEGA trajectory type.

Table 4-5

1995 URANUS ORBITER/PROBE MISSION(1) FLIGHT TIME (YRS) COMPARISON

Flight Mode/ Orbit Capture Mode(2)	Launch Vehicle/Upper Stage: Shuttle/Centaur(G')	OOA Centaur(G')	OOA Centaur(G')/ Centaur(G)
J/U BAL/E-S	12.3	10.2	8.3
J/U BAL/S-S	12.4	10.0	8.0
J/U BAL/A-C	--	--	6.0
J/U ΔVEGA/E-S	11.5	10.7	10.0
J/U ΔVEGA/S-S	10.8	10.0	9.4
J/U ΔVEGA/A-C	8.7	7.4	6.5

(1) 240 kg probe, 3 R_U x 60 day orbit

(2) Required Payload into Orbit (kg):

Chemical Capture	816
Aerocapture	1056

The need for the stacked Centaur concept to enable the aerocapture option for ballistic missions can be seen in Figure 4-8 which demonstrates that at the longer flight times (and corresponding lower approach velocities) the weight of the aerocapture system becomes a penalty as compared to the standard chemical retro options. The 1993 ΔVEGA performance is shown in Figure 4-9. In the case of the ΔVEGA trajectory type, note the substantial reduction in flight time due to aerocapture and also to the fact that the flight times for the chemical retropropulsion option are comparable to the direct ballistic case.

4.4.2 Post-2000 Multiprobe Mission

The performance for the three-probe mission for the complete spectrum of flight mode/trajectory type and launch vehicle/retro options is summarized in Table 4-6. A -- indicates data are not available, while ( ) indicates no mission is possible at a reasonable flight time. The data are summarized for the three probe missions because, as will be shown, the effect of total probe

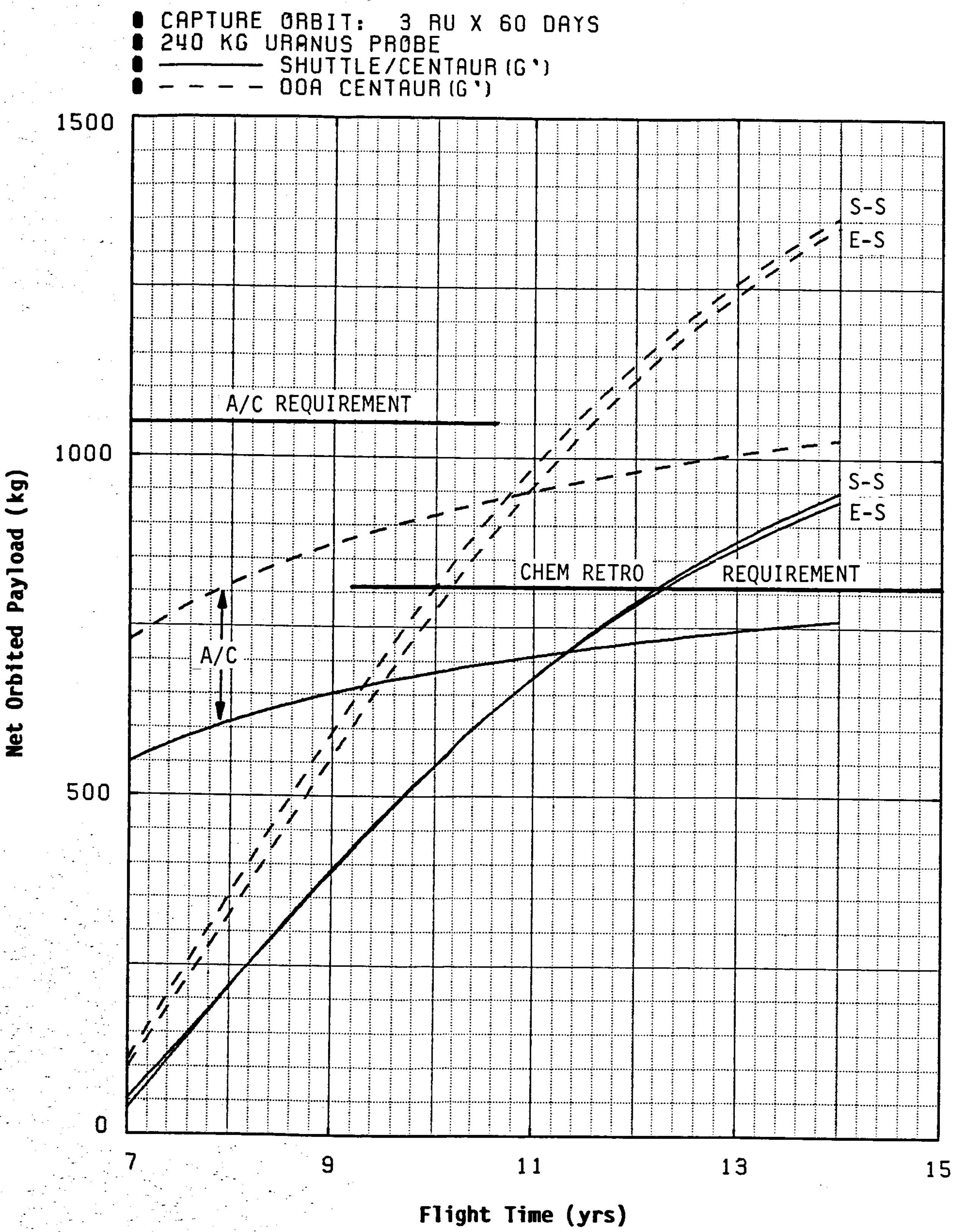

Figure 4-8. Uranus Orbiter/Probe Mission J/U Ballistic (1995) Performance

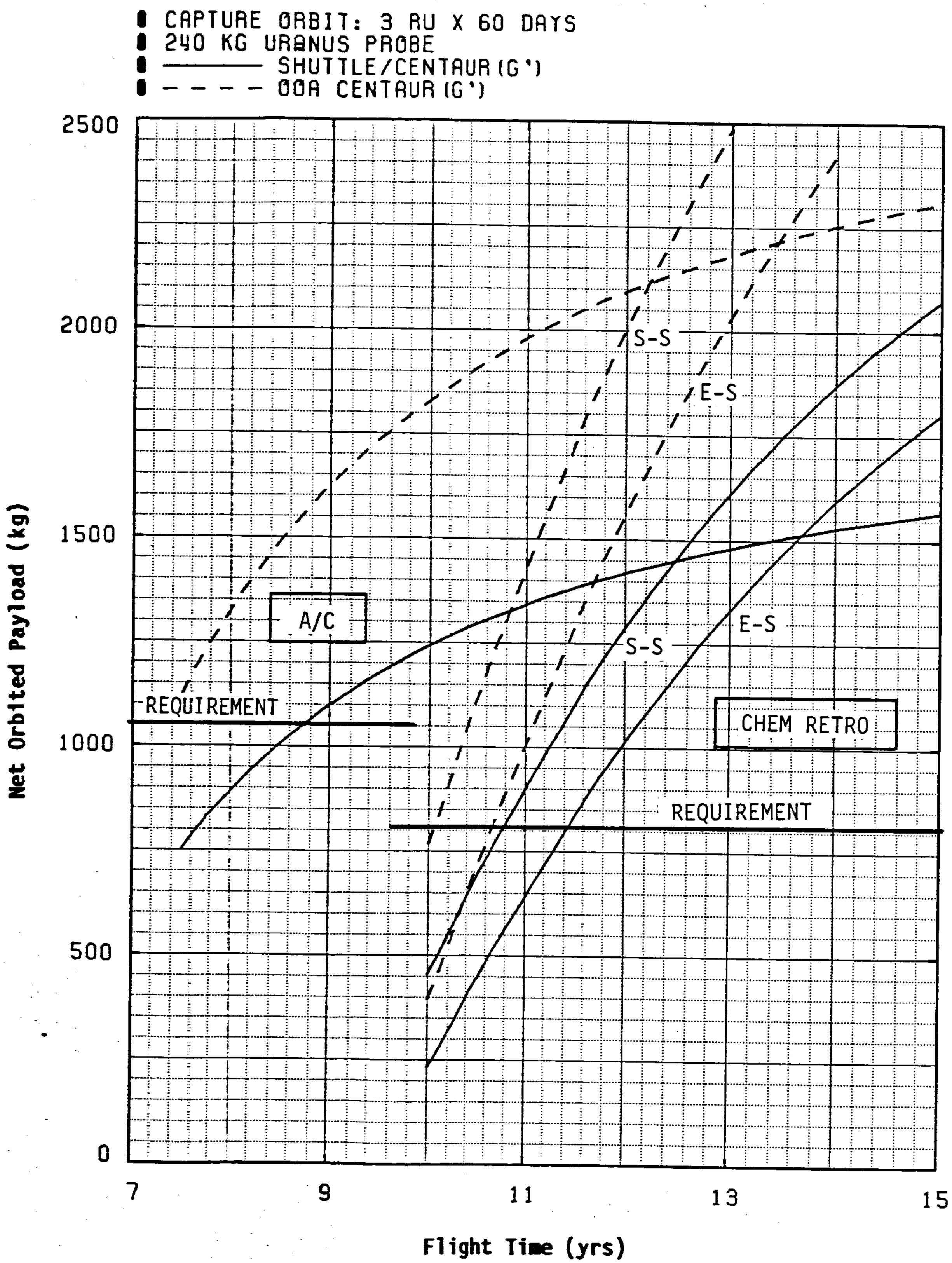

Figure 4-9. Uranus Orbiter/Multiprobe Mission J/U ΔVEGA Ballistic (1993) Performance

Table 4-6

URANUS MULTIPROBE MISSION[1] FLIGHT TIME (YRS) COMPARISON

Flight Mode/ Orbit Capture Mode[3]	Launch Vehicle/Upper Stage: Shuttle/ Centaur(G')	OOA Centaur(G')	OOA Centaur(G')/ Centaur(G)	OTV(4-R) OTV(2-E)
NEP/Spiral 9.8[2]	--	8.4	--	5.9
SEEGA/E-S	10.1	--	--	--
SEEGA/S-S	9.7	--	--	--
SEEGA/A-C	10.3	--	--	--
J/U SEEGA/E-S	11.5	--	--	--
J/U SEEGA/S-S	11.3	--	--	--
J/U SEEGA/A-C	7.4	--	--	--
J/U BAL/E-S	( )	11.1	9.0	8.4
J/U BAL/S-S	( )	10.8	8.7	8.1
J/U BAL/A-C	( )	( )	6.6	5.2
J/U ΔVEGA/E-S	12.1	10.8	10.2	9.9
J/U ΔVEGA/S-S	11.3	10.4	9.6	9.4
J/U ΔVEGA/A-C	( )	8.1	7.0	6.6

(1) Three 347.2 kg probes, 3 R_U x 60^d orbit

(2) Earth Escape Spiral

(3) Required Payload into Orbit (kg):

NEP	2500
SEEGA or BAL (A-C)	1880
SEEGA or BAL (Chem)	840

mass on trip time is not significant when compared to the other factors. In general, aerocapture results in the shortest trip time for all the flight modes, and the ΔVEGA trajectory adds at least one year to the ballistic flight times, but enables the mission with the less capable launch vehicles.

Representative performance curves for the J/U ballistic opportunity in 2007 and the corresponding ΔVEGA in 2005 are shown in Figure 4-10. The curves are plotted for the stacked Centaur concept and Earth-storable chemical retropropulsion and aerocapture. The most striking result is the substantial

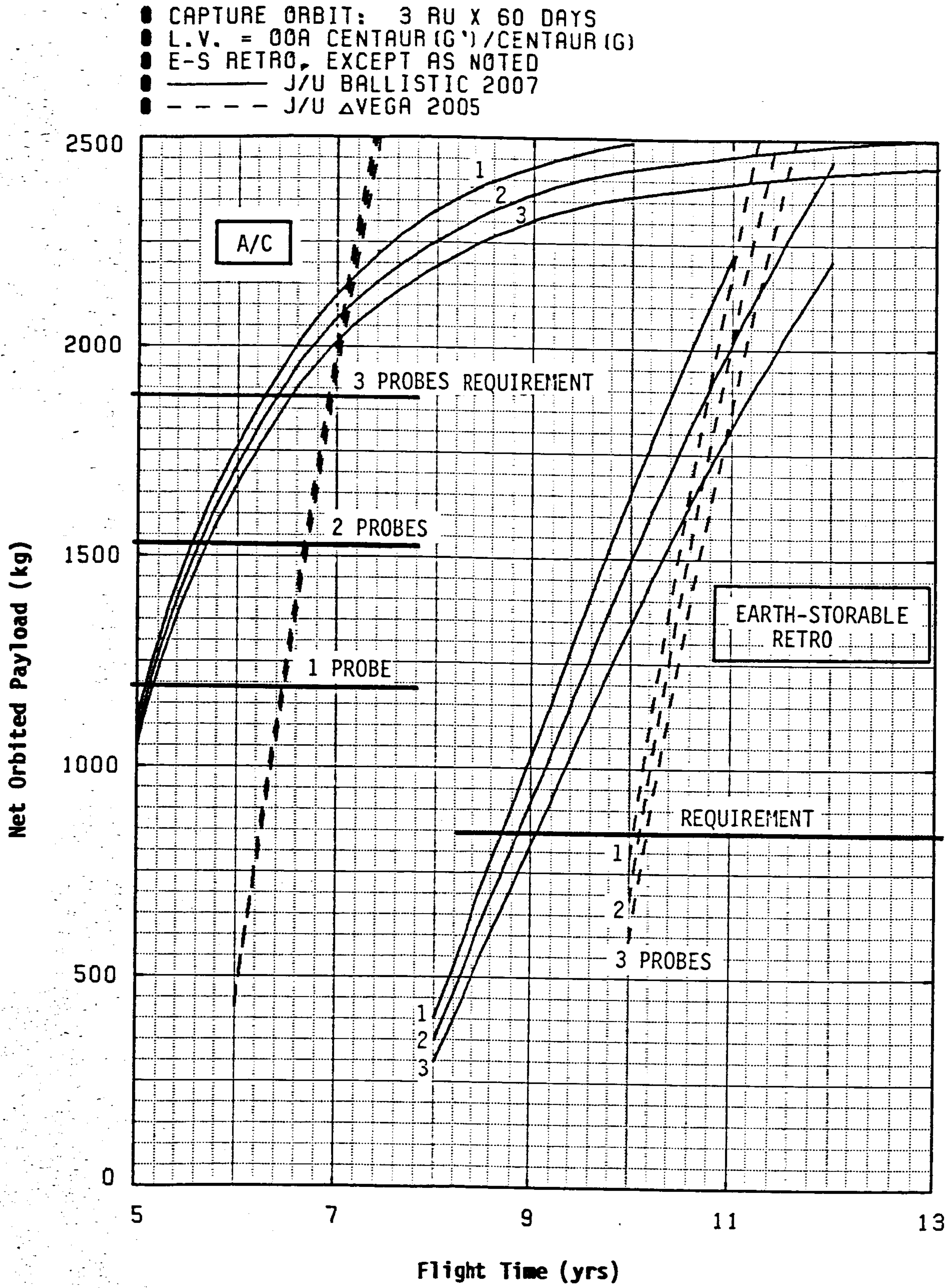

Figure 4-10. Uranus Orbiter/Multiprobe Mission Ballistic Performance

decrease in flight time due to the use of aerocapture. Secondly, note the rather secondary effect of the number of probes on the total mission trip time; e.g., for the ΔVEGA missions these probes have hardly any effect.

Representative SEEGA performance is shown in Figure 4-11, also for the stacked Centaur and Earth-storable retropropulsion and aerocapture. The 2^+ SEEGA assumes launch in any year, while the J/U SEEGA launches in 2005. Due to the increased Uranus approach velocities for the J/U SEEGA trajectory, the performance is lower for the case of chemical retro orbit insertion than for the direct 2^+ SEEGA. However, when aerocapture, which is less sensitive to approach velocity, is used, the effect of the lower launch energy dominates and flight times are reduced by as much as 35%.

The NEP results are plotted in Figure 4-12. A total trip time of nearly 10 years is required for the three-probe mission using an Earth-escape spiral. (This includes 1-1/2 years of spiral time). However, this flight time is reduced by 40% to slightly less than six years by launching the spacecraft and NEP system to greater than escape energy on-board an OTV(4-R)/OTV(2-E).

4.5 Neptune

The flight time comparison for the Neptune Orbiter/Dual Probe Mission is summarized in Table 4-7. As with the Uranus case, aerocapture gives the shortest flight time for any flight mode or trajectory type, while the ΔVEGA adds at least one year to the total mission duration, but enables the mission with the less capable launch vehicles.

Representative performance for the J/N ballistic opportunity in 2006 is shown in Figure 4-13. Aerocapture technology can reduce the flight time by 35 to 40 percent. The performance for the corresponding ΔVEGA in 2004 is shown in Figure 4-14. Again, aerocapture offers substantial savings in flight time over chemical retro orbit insertion.

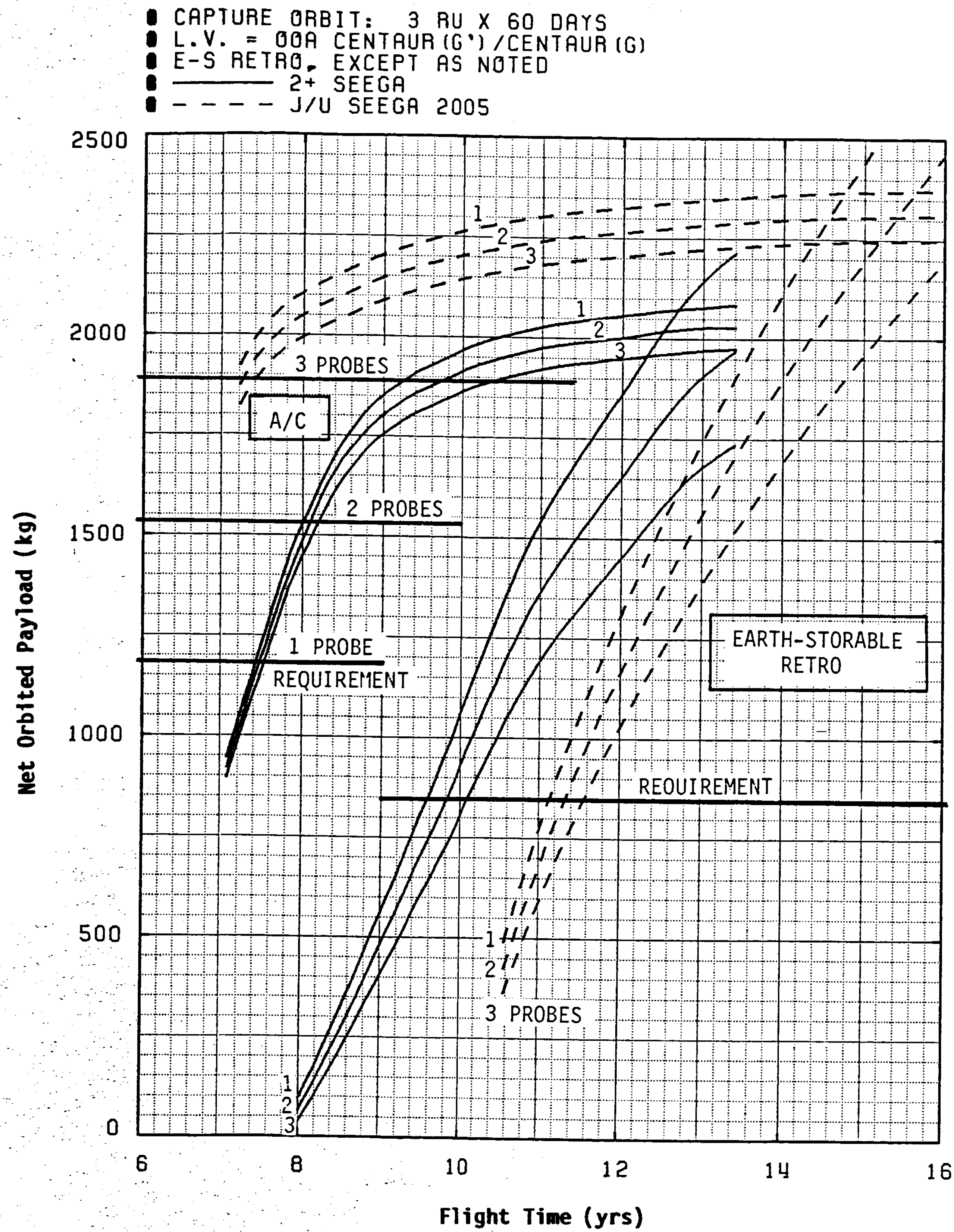

Figure 4-11. Uranus Orbiter/Multiprobe Mission SEEGA Performance

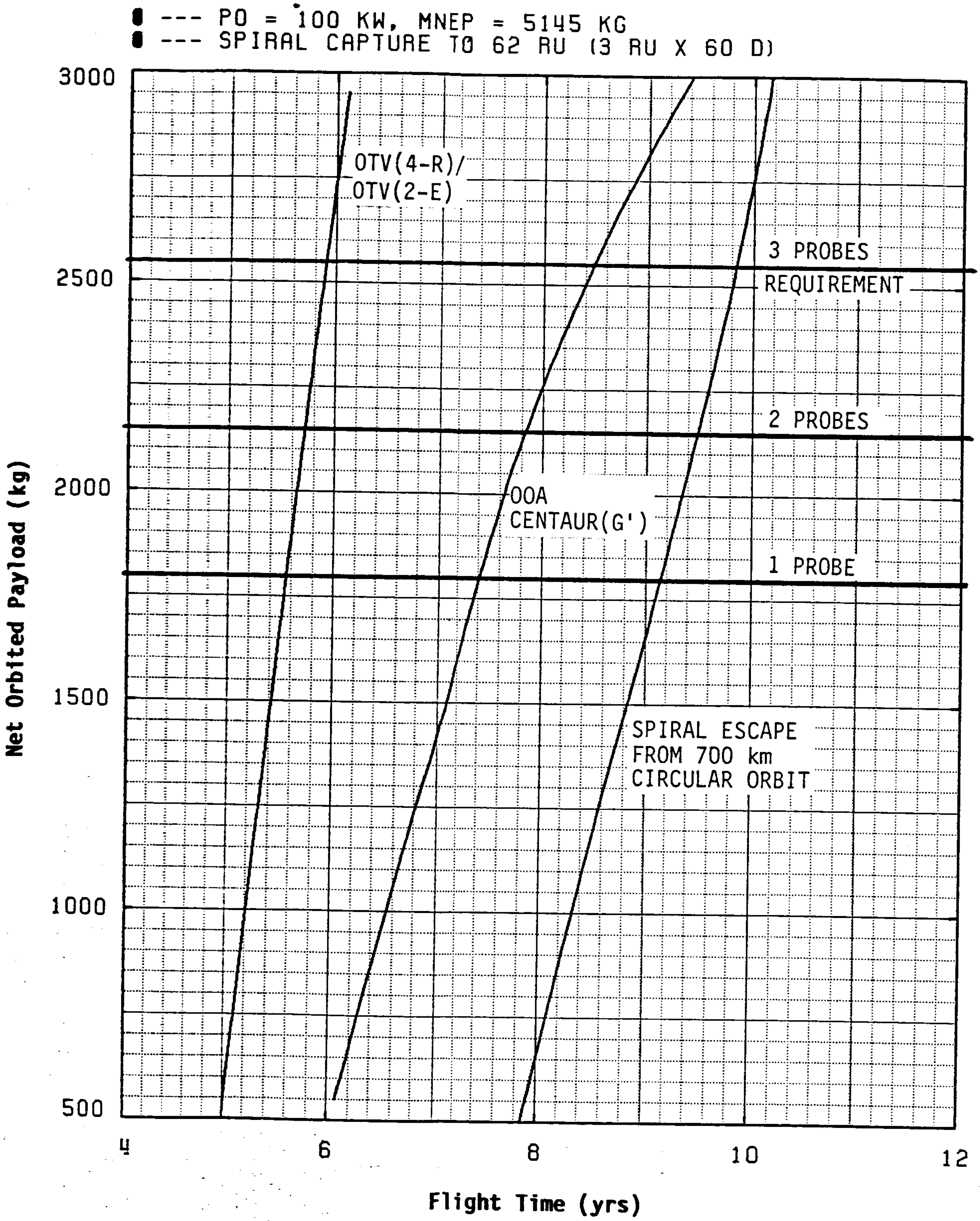

Figure 4-12. Uranus Orbiter/Multiprobe Mission NEP Performance

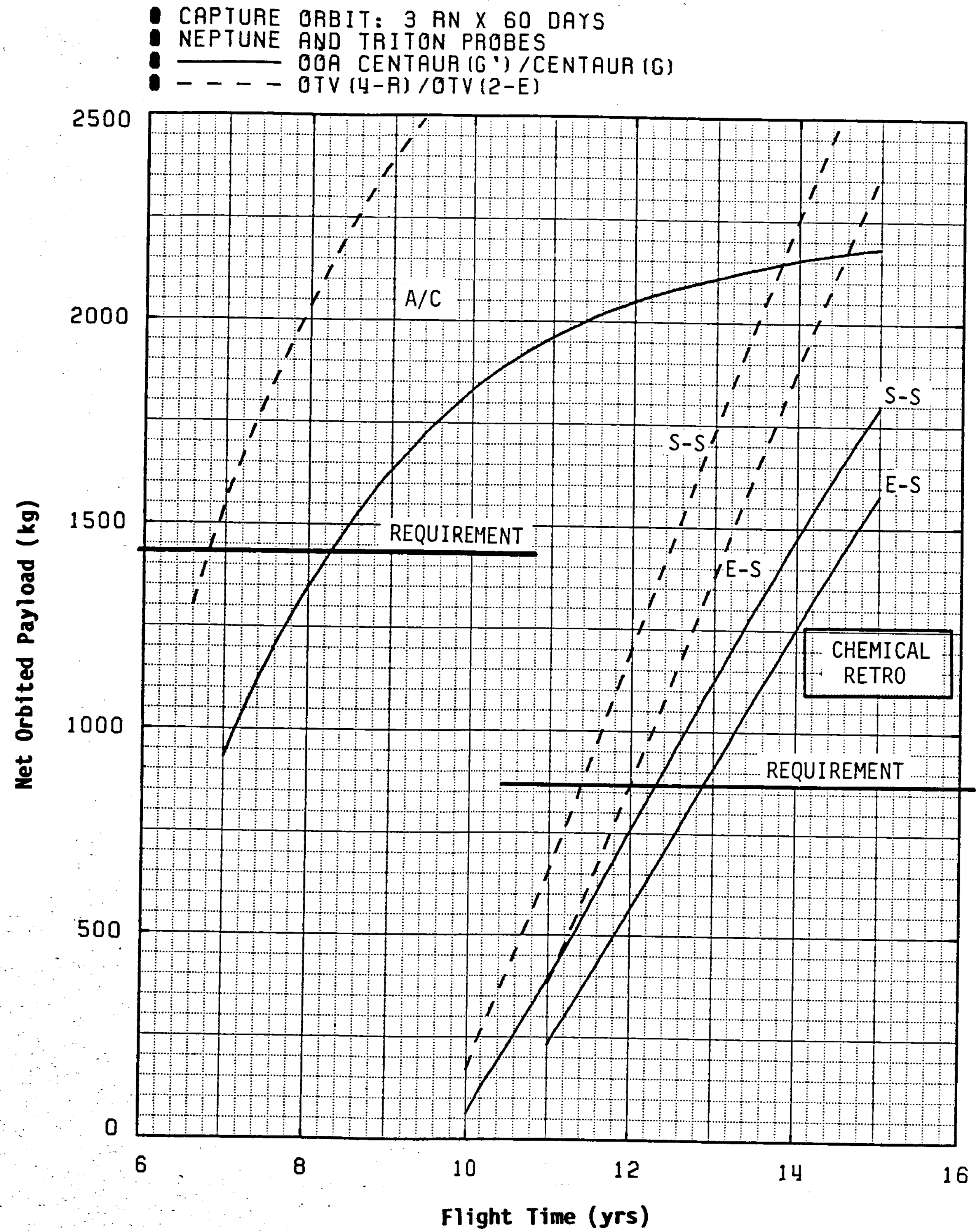

Figure 4-13. Neptune Orbiter/Dual Probe Mission J/N Ballistic (2006) Performance

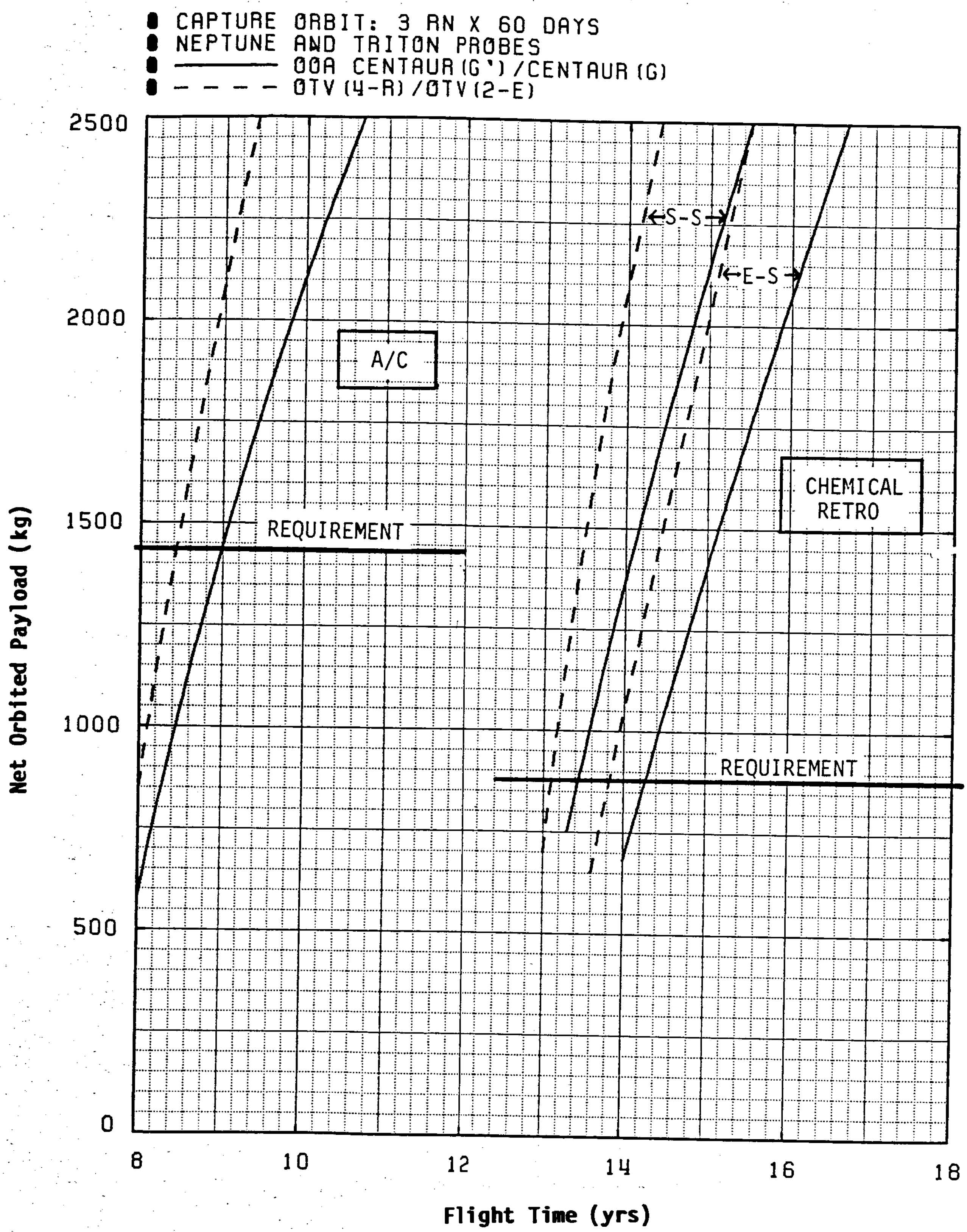

Figure 4-14. Neptune Orbiter/Dual Probe Mission J/N ΔVEGA Ballistic (2004) Performance

Table 4-7

NEPTUNE ORBITER/DUAL PROBE MISSION[1] FLIGHT TIME (YRS) COMPARISON

	Launch Vehicle/Upper Stage			
Flight Mode/ Orbit Capture Mode[3]	Shuttle/ Centaur(G')	OOA Centaur(G')	OOA Centaur(G')/ Centaur(G)	OTV(4-R)/ OTV(2-E)
NEP/Spiral 11.9[2]	--	11.2	--	7.9
SEEGA/E-S	14.6	--	--	--
SEEGA/S-S	14.1	--	--	--
SEEGA/A-C	12.6	--	--	--
J/N SEEGA/E-S	15.0	--	--	--
J/N SEEGA/S-S	14.4	--	--	--
J/N SEEGA/A-C	8.6	--	--	--
J/N BAL/E-S	( )	( )	12.9	12.0
J/N BAL/S-S	( )	( )	12.3	11.4
J/N BAL/A-C	( )	( )	8.3	6.8
J/N ΔVEGA/E-S	16.9	15.4	14.2	13.7
J/N ΔVEGA/S-S	15.8	14.4	13.5	13.1
J/N ΔVEGA/A-C	14.5	10.7	9.0	8.4

(1) Neptune & Triton probe, 3 R_N x 60^d orbit

(2) Earth-Escape Spiral

(3) Required Payload into Orbit (kg):

NEP	2050
SEEGA or BAL(A-C)	1430
SEEGA or BAL(Chem)	870

Representative SEEGA performance is displayed in Figure 4-15, comparing a 2^+ SEEGA launched in any year with the J/N SEEGA opportunity in 2005. As with Uranus, the 2^+ SEEGA outperforms the J/N SEEGA when chemical retro stages are employed, but the J/N SEEGA with aerocapture is by far the mode with the shortest flight time, representing over 60% in flight time savings (6.4 years) over the J/N SEEGA with Earth-storable retro.

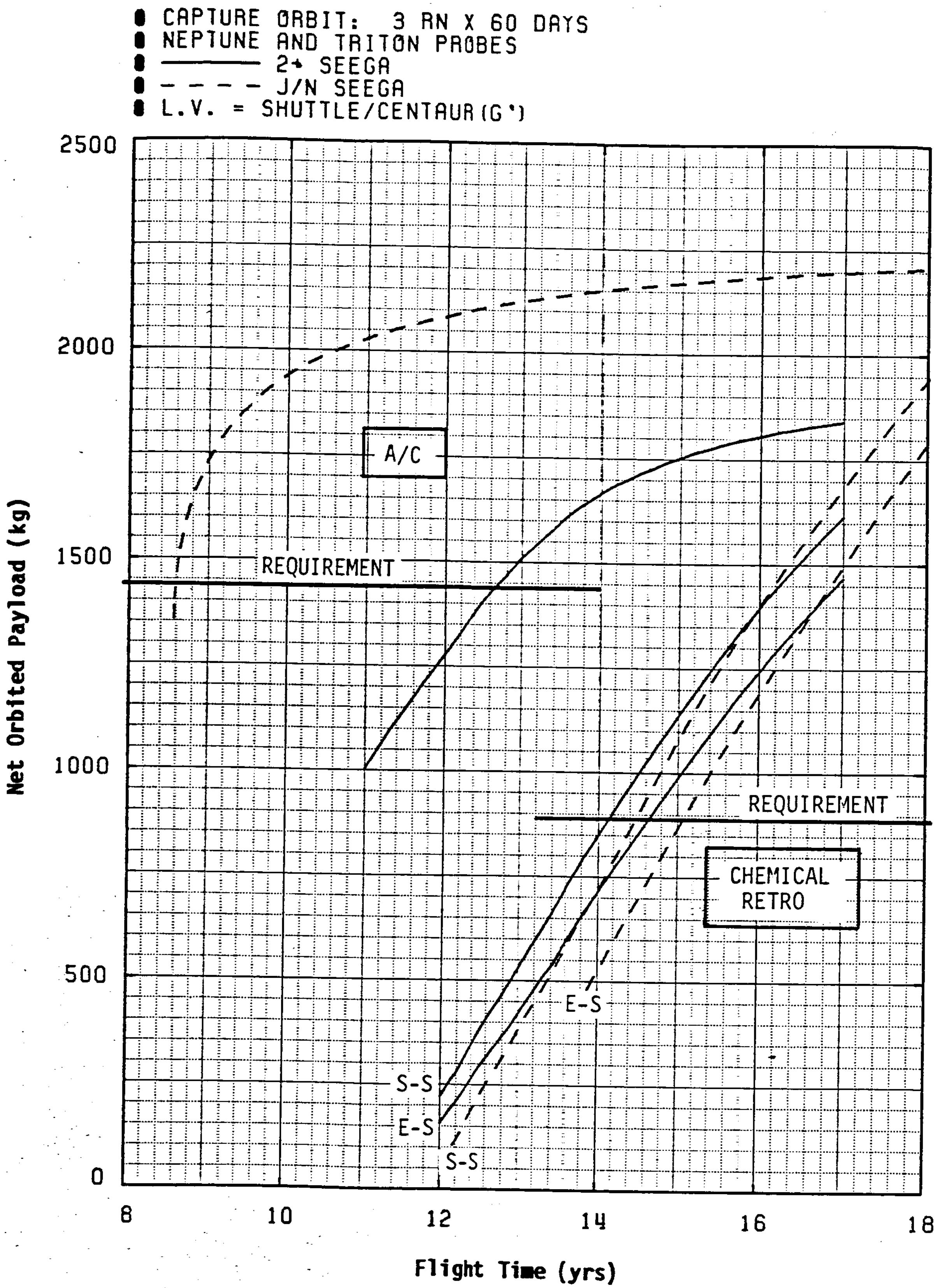

Figure 4-15. Neptune Orbiter/Dual Probe Mission SEEGA Performance

Finally, the NEP performance is shown in Figure 4-16. A reduction in the mission trip time of over four years is possible by launching the spacecraft/ NEP system to $C_3 > 0$ on the OTV(4-R)/OTV(2-E), as opposed to an Earth-escape spiral.

4.6 Pluto

As mentioned previously, the only delivery option considered for the Pluto Orbiter/Lander and Charon Lander mission is the NEP flight mode due to the lengthy flight times and substantial energy requirements to reach this outermost target. Also, any atmosphere that may exist on Pluto may not support aerocapture, and the high approach velocities, coupled with Pluto's small gravitational field, would result in prohibitively massive chemical retro stages. Mission performance is shown in Figure 4-17 for a spiral capture to Charon's orbital radius (13.1 Pluto radii).

A total trip time of over 11 years is necessary to deliver the required 1600 kg of payload into orbit if an Earth-escape spiral is assumed. (This includes the 1-1/2 years of escape spiral time.) Substantial flight time reductions can be realized if the spacecraft with NEP system is launched to greater than escape energy. As shown in Figure 4-17, this trip time is reduced to slightly over 7-1/2 years when launched using the OTV(4-R)/ OTV(2-E).

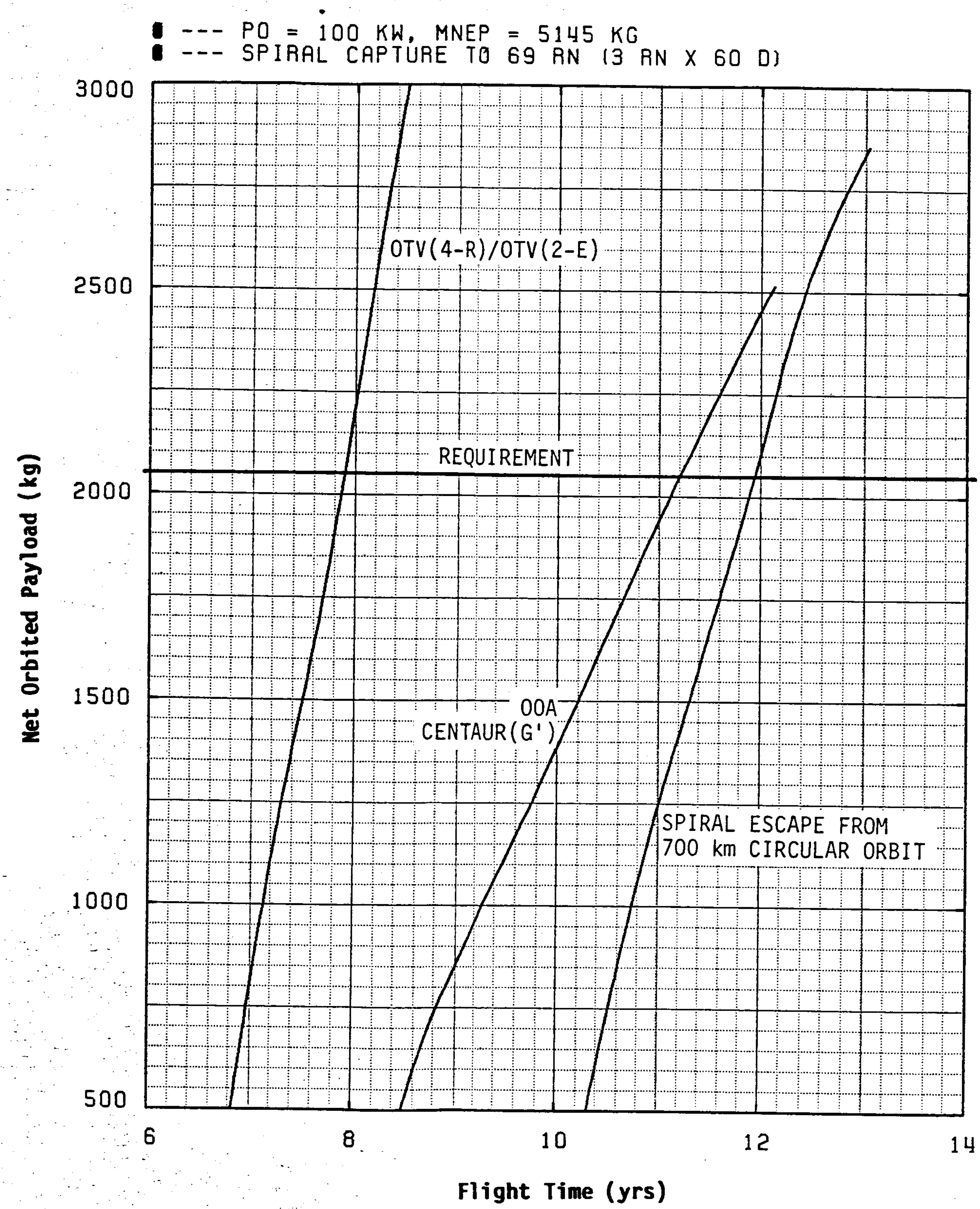

Figure 4-16. Neptune Orbiter/Dual Probe Mission NEP Performance

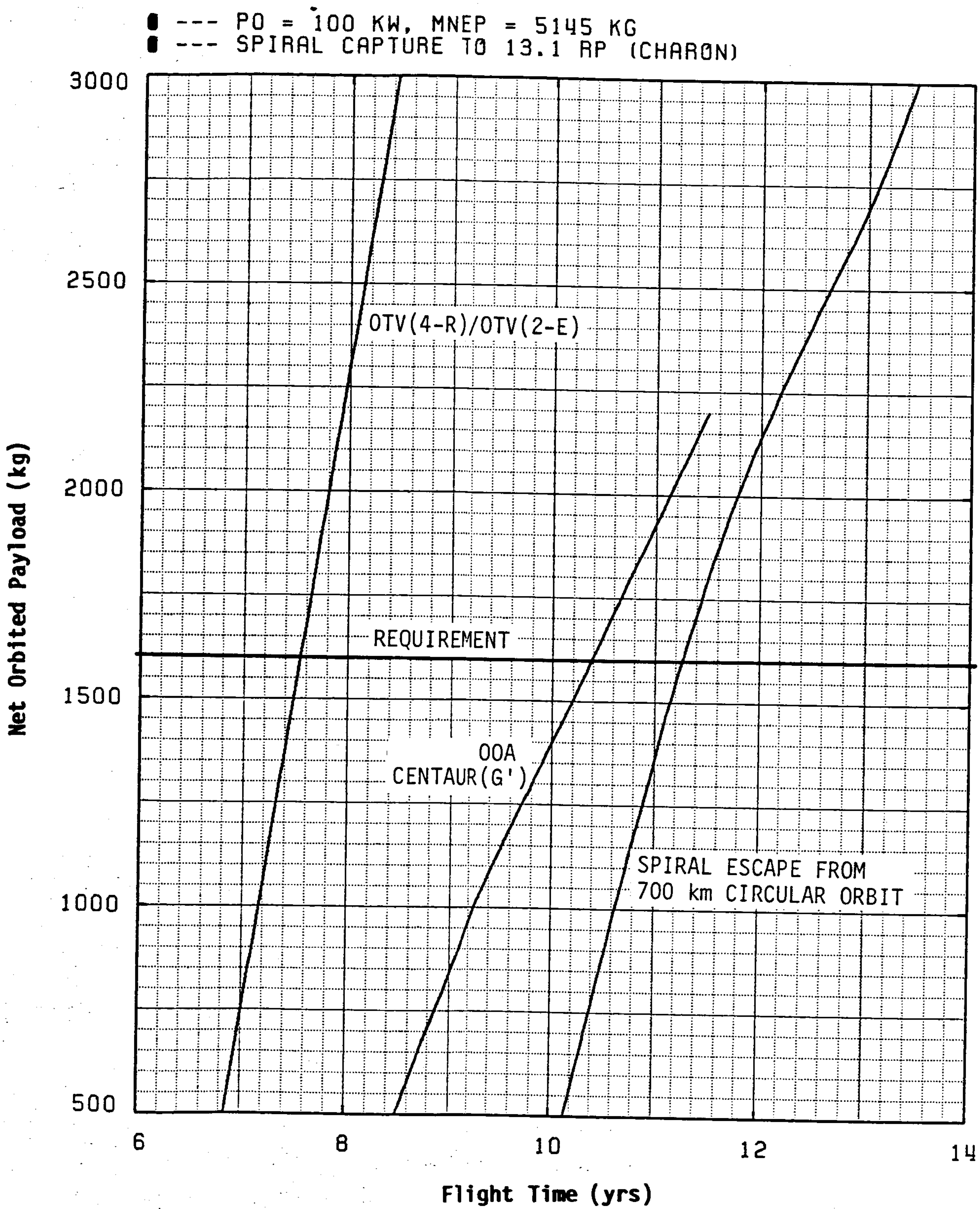

Figure 4-17. Pluto Orbiter/Lander and Charon Lander Mission NEP Performance

5. CANDIDATE MISSION DESCRIPTIONS

5.1 Introduction

Information presented in this section summarizes the important features of each of the missions discussed in the previous three sections of this report. All mission descriptions use the same format consisting of the following three major components:

1. Science Objectives and Instrument Payload;
2. Mission Performance; and
3. Cost Estimate.

The details of each of these components are discussed below.

The science objectives for each mission (already listed in Section 2 of this report) are briefly restated at the beginning of each description. In addition, any special constraints imposed on the mission or deviations from the original science objectives (resulting from subsequent analysis) are noted here. This is followed by a table of required masses for the science instruments needed to meet these objectives.

A mission performance section then describes major events in the outbound flight and the mass performance of the overall mission. The first part of this section indicates the flight mode (ballistic, SEEGA or NEP), type of trajectory (direct, swingby, or Δ VEGA) and capture mode (chemical retropropulsion, aerocapture or NEP spiral). Specific trajectory data are then provided including: launch energy (C_3), declination of launch asymptote (DLA), assumed launch date for ballistic and SEEGA trajectories, and total trip time from Earth to the target. These trajectory data have been used with a specific launch vehicle to estimate an injected mass margin, which makes up the last part of this section.

The final section of each mission description consists of an estimated cost to carry out that mission (Ref. 30). This estimate includes the cost of the following hardware elements and project activities:

Mission Vehicles (orbiters, penetrators, probes and landers): science hardware development; engineering hardware development; system design, integration and test; project management.

RTG: radioisotope thermoelectric generator power source.

Flight Vehicle Systems (aerocapture vehicle, NEP stage): same as mission vehicles but without science costs.

Vehicle Integration: integration of multiple vehicle flight stacks.

Launch + 30 days: ground data system and ground system development; ETR operations.

Program Management: top level, overall management and contract monitoring (technical and administrative).

Contingency: 30% of all estimated costs to this point except the 1995 Uranus Orbiter/Probe which assumes 20% due to Mariner Mark II heritage.

Flight Operations: cruise and encounter operations.

Data Analysis: analysis of returned science data.

Transportation: cost for Earth-escape upper stage(s), on-orbit assembly (if required) and STS cost to place the spacecraft and expendables in orbit.

Several assumptions have been incorporated into each of the cost estimates. These include:

1. All orbiter and carrier spacecraft are developed in-house by JPL; all other hardware elements are built via major system contracting;
2. No international cooperation; and
3. All costs listed in FY 1986 dollars.

The cost estimates for flight operations are based on average rates for cruise and encounter derived from cost projections for the Galileo Mission. Besides regular operations activities, the estimates also include management, mission design and science team support. Because the missions considered in this study have not been integrated into a multi-mission plan, flight operations for each mission are costed in a stand-alone mode. Flight operations costs are therefore most likely conservative.

Other than certain ambitious planetary missions, practical applications of nuclear electric propulsion, which may help drive its development, have not yet been identified. In the worst case, a planetary mission requiring NEP would have to pay the full development cost, which in all likelihood would be prohibitive. Since a prorated cost sharing of NEP development cannot be determined, only an estimate of the recurring (unit) cost of a NEP stage is included for those missions using this propulsion system. Thus the cost assumption is most likely optimistic.

The addition of a cost estimate to the information generated in previous sections marks the completion of each mission analysis. Finally, if any new technology issues were identified during this investigation, recommendations for specific additional studies are made.

5.2 Jupiter Inner Magnetosphere/Polar Orbiter

5.2.1 Science Objectives and Payload

A close polar orbiter at Jupiter will allow detailed cross-sectional measurements to be made of the intense inner magnetosphere and charged particle radiation belts. This vehicle could also observe Io's role in the development of auroral activity in Jupiter's atmosphere and the Io plasma torus. Among the specific science objectives related to in situ and remote sensing measurements and orbiter tracking are:

1. Determine the density, composition and energy of magnetospheric particles;
2. Observe the large-scale structure and rotation of the magnetosphere;
3. Observe the time-dependent phenomena of the magnetosphere as related to Io, other satellites, orbiting gases and plasmas;
4. Determine the nature of auroral activity and the ionosphere;
5. Determine the nature of electromagnetic radio emission; and
6. Observe the harmonics of Jupiter's gravitational field.

A candidate science payload based on Galileo and Pioneer Venus Orbiter heritage is listed in Table 5-1. The payload consists of 11 instruments/ experiments, nine of which are of the in situ particles and fields type and two of which are remote sensing instruments mainly in the UV spectral range. In addition, radio Doppler tracking of the orbiter from Earth provides opportunity for atmospheric occultations and gravity field determination. The total science payload weighs 69 kg, requires 56 W of power, and generates about 3000 bits/sec of data if all instruments are operating simultaneously. A spin-stabilized spacecraft is envisioned for this mission application, because of its minimum operational complexity and mass. Table 5-1 lists the orbiter mass by subsystem exclusive of the propulsion subsystem. The net orbiter mass is estimated to be 550 kg including an allowance for added radiation shielding of sensitive electronics and a 10% contingency. Due to the intense radiation environment, the electronics will probably have to be more radiation-tolerant than the Galileo design, even with added shielding.

Table 5-1

JUPITER INNER MAGNETOSPHERE/POLAR ORBITER - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
TRI-AXIAL MAGNETOMETER	7	3.7	240
PLASMA DETECTOR	12	7.2	600
PLASMA WAVE	6	3.8	240
ENERGETIC PARTICLE DETECTOR	9	7.4	912
COSMIC RAY DETECTOR	7	5.4	240
DUST DETECTOR	4	1.7	12
ELECTRON TEMPERATURE PROBE	6	3.0	256
ION MASS SPECTROMETER	3	1.5	65
NEUTRAL MASS SPECTROMETER	7	15.0	256
IMAGING PHOTOPOLARIMETER	5	5.4	33
UV SPECTROMETER	3	2.0	128
TOTAL	69	56.1	

- **ORBITER SUBSYSTEMS (EXCLUDING PROPULSION)**

SCIENCE	69
STRUCTURE AND DEVICES	139
THERMAL, CABLING AND PYRO	52
ATTITUDE AND ARTICULATION CONTROL	20
TELECOMMUNICATIONS	30
ANTENNAS	8
COMMAND AND DATA	50
RTG	77
POWER SUPPORT	35
RADIATION SHIELDING	20
CONTINGENCY (10%)	50
TOTAL	550 kg

5.2.2 Mission Performance

Mission performance is summarized in Table 5-2. A direct ballistic transfer from Earth to Jupiter of 800 days' duration with a launch in the year 2000 is assumed. The nominal operational orbit places perijove 1000 km above the cloud tops with apojove at 12 R_J; the period of this orbit is two Earth days. Initial perijove latitude lies near the equator but then moves toward the polar region (north or south depending on initial targeting) at the rate of 1.2 degrees/day due to Jupiter's oblateness perturbation. The spacecraft's total V budget is 2.83 km/sec and it is implemented by Earth-storable retropropulsion at a specific impulse of 315 sec. The Jupiter Inner Magnetosphere/Polar Orbiter mission can be launched by the Shuttle/Centaur(G') with a comfortable injected mass margin of 310 kg.

5.2.3 Mission Cost

The cost estimate for this mission is summarized in Table 5-3. Total project cost for hardware development, launch, flight operations, and data analysis is $461M which includes a liberal 30% contingency through vehicle launch. Transportation costs to NASA for the Shuttle and Centaur(G') add another $148M, for a total estimated program cost of $609M.

5.2.4 Additional Study Recommendation

A more detailed study of this mission is clearly needed. Among the key issues related to mission feasibility and design choices are: (1) assurance that the science instruments and spacecraft subsystems will function reliably in the intense radiation environment; and (2) possible requirements for a more capable (and heavier) axis-stabilized spacecraft bus (e.g. Mariner Mark II design heritage) for purposes of remote sensing instrument pointing including higher quality CCD imaging for both science and optical navigation of close Jupiter and (possibly) Io encounters.

Table 5-2

JUPITER INNER MAGNETOSPHERE/POLAR ORBITER - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... DIRECT

  CAPTURE MODE .... EARTH-STORABLE RETRO

  ORBIT .... 1.014 x 12 R_J, POLAR

- **TRAJECTORY**

  C_3 $(km/sec)^2$ .... 86.7 LAUNCH DATE 8/5/2000

  DLA (deg) .... 30.0 TOTAL TRIP TIME (yrs) .. 2.2

  $V_{\infty J}$ (km/sec) .... 6.2 SPACECRAFT ΔV (km/sec) . 2.83 km/sec

- **MASS PERFORMANCE** L.V. = SHUTTLE/CENTAUR(G')

  ORBITER .... 550 kg

  CHEMICAL RETROPROPULSION 1383
  PROPELLANT 1160
  INERTS 223

  INITIAL MASS 1933
  L.V. ADAPTER 97

  INJECTED MASS REQUIRED 2030

  INJECTED MASS CAPABILITY 2340
  INJECTED MASS MARGIN 310

Table 5-3

JUPITER INNER MAGNETOSPHERE/POLAR ORBITER - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$235M
RTG	25
LAUNCH + 30 DAYS OPERATIONS	16
PROGRAM MANAGEMENT	5
CONTINGENCY (30%)	83
TOTAL DEVELOPMENT	$361M
FLIGHT OPERATIONS	80
DATA ANALYSIS	20
TOTAL MISSION OPERATIONS	$100M
SHUTTLE LAUNCH	71
CENTAUR	77
TOTAL TRANSPORTATION	$148M
TOTAL PROGRAM COST	$609M

5.3 Jupiter Deep Probe/Multiprobe

5.3.1 Science Objectives and Payload

The Galileo Probe will be the first probe to make direct measurements of the Jovian atmosphere. However, only one small region will be examined by this vehicle in one of the most turbulent and complex atmospheres in the solar system. The proposed mission will make a more global study of the atmosphere through the use of four atmospheric probes. Three of these probes (the multiprobe) will be limited to relatively low pressures while the single remaining probe (the deep probe) will be designed to withstand much higher pressure levels. The multiprobe science objectives focus on characterizing the dynamics, structure and composition of the Jovian atmosphere down to the 20 bar pressure level at three widely separated locations. All three of these locations will be in the southern hemisphere with one of the probes specifically targeted for the Great Red Spot. The science objectives for the deep probe are basically the same as those of the multiprobe but will be carried out to the 1000 bar pressure level. This deep probe will be released from the carrier spacecraft on approach and will be targeted for entry at the equator. A strawman science payload using instruments similar to those of the Galileo, Pioneer Venus and proposed Titan probes is shown in Table 5-4. The deep probe contains six instruments, one of which is a remote sensing device while the remaining five are direct sensing instruments. The total science mass for this probe is 28.5 kg which will use a maximum of 99 W of power and will transmit no more than 144 bits/sec of data. A relay communications probe will be required for this mission to maintain an adequate data rate to the orbiter. The multiprobes each carry 11 instruments, four of which make up a pre-entry science package. Of these 11 instruments, 10 are of the direct sensing type and the remaining one is a remote sensing instrument. The total science payload mass for each probe will be 54.5 kg of which 19.4 kg is in the pre-entry package. Once the pre-entry package has been released, each probe will require 116 W to operate all instruments and will transmit a maximum of 174 bits/sec. A spin-stabilized carrier bus will be used to deliver these probes individually to their designated target. The bus will then serve as a communications relay to Earth. No science will be carried out by the bus. Table 5-4 lists the mass values by subsystem for the probe carrier and each

Table 5-4

JUPITER DEEP PROBE/MULTIPROBE - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)	DEEP PROBE	MULTI-PROBE
ATMOSPHERIC STRUCTURE	3.8	13	18	X	X
HELIUM ABUNDANCE DETECTOR	1.4	5.5	4	X	X
NEPHELOMETER	4.4	13.5	10	X	
CLOUD PARTICLE SIZE	4.4	20	30		X
NET FLUX RADIOMETER	2.7	11.8	32		X
NEUTRAL MASS SPECTROMETER	12.3	42	32	X	X
LIGHTNING + RADIO DETECTOR/ ENERGETIC PARTICLE INSTRUMENT	2.5	3.3	8		X
PRE-ENTRY SCIENCE	23.8	22	36		X
MICROWAVE RADIOMETER	3	5	30	X	
GAS CHROMATOGRAPH	3.6	20	50	X	X

- **VEHICLE SUBSYSTEMS (EXCLUDING PROPULSION)**

	BUS	DEEP PROBE	MULTIPROBE
SCIENCE	0	28	55
STRUCTURE AND DEVICES	252	102	35
THERMAL, CABLING AND PYRO	60	15	9
ATTITUDE AND ARTICULATION CONTROL	39	0	0
TELECOMMUNICATIONS	41	11	11
ANTENNAS	20	2	2
COMMAND AND DATA	53	17	17
POWER SOURCE AND PROCESSING	112	20	13
ENTRY SHELL	0	371	219
RELAY PROBE	0	30	0
CONTINGENCY (10%)	58	60	---
TOTAL	635 kg	656 kg	361 kg

probe type. The net mass of the carrier and the four probes (without including any propellant) is seen to be 2291 kg.

5.3.2 Mission Performance

Trajectory and mass performance information for this mission has been summarized in Table 5-5. This mission will use a direct ballistic trajectory which would be launched in August of 2000 and requires approximately 700 days to reach Jupiter. After release of the deep probe, the carrier bus will be captured into a 90-day elliptical orbit with a perijove distance of 5.3 Jupiter radii. The spacecraft will change orbit inclination by a total of 50 degrees to allow the shallow probes to be released into different regions of the atmosphere on successive orbits. To accomplish the orbit capture and inclination change, an Earth-storable propulsion system (I_{sp} = 315 sec) with a mass of 1868 kg will be used. The entire spacecraft stack has a mass of 4367 kg which will be launched from low-Earth orbit by a Centaur(G')/Centaur(G) upper stage combination.

5.3.3 Mission Cost

The estimated cost to carry out this mission is shown in Table 5-6. Total project cost for hardware development, launch, and flight operations (including a 30 percent contingency through launch) is $1095M in FY'86 dollars. Transportation costs for the Space Shuttle and upper stages will add $306M to the project costs yielding a total mission cost of $1401M.

5.3.4 Additional Study Recommendation

The ability to keep any of the probes functioning for longer periods of time is highly desirable for this type of mission. Balloons have been shown to be unfeasible in Jupiter's atmosphere. An alternative approach would be the use of lifting devices such as inflatable airfoils. Such devices will require an engine for propulsion or favorable wind gradients to maintain altitude. This concept will require further study to assess its feasibility.

Table 5-5

JUPITER DEEP PROBE/MULTIPROBE - PERFORMANCE SUMMARY

- **DESCRIPTION**

FLIGHT MODE	BALLISTIC
TRAJECTORY TYPE	DIRECT
CAPTURE MODE	EARTH-STORABLE RETRO

- **TRAJECTORY**

C_3 (km/sec)^2	88.7	LAUNCH DATE	8/2/2000
DLA (deg)	22.4	TOTAL TRIP TIME (yrs)	1.9

- **MASS PERFORMANCE** L.V. = 00A CENTAUR(G')/CENTAUR(G)

ORBITER		635 kg
LARGE PROBE		656
SMALL PROBES (3)		1083
CHEMICAL RETROPROPULSION		1868
PROPELLANT	1587	
INERTS	281	
INITIAL MASS		4242
L.V. ADAPTER		208
INJECTED MASS REQUIRED		4450
INJECTED MASS CAPABILITY		5442
INJECTED MASS MARGIN		992

Table 5-6

JUPITER DEEP PROBE/MULTIPROBE - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 266M
MULTIPROBES (3)	215
DEEP PROBE	128
RTG	25
VEHICLE INTEGRATION	29
LAUNCH + 30 DAYS OPERATIONS	44
PROGRAM MANAGEMENT	58
CONTINGENCY (30%)	230
TOTAL DEVELOPMENT	$ 995M
FLIGHT OPERATIONS	80
DATA ANALYSIS	20
TOTAL MISSION OPERATIONS	$ 100M
SHUTTLE LAUNCHES (2)	142
CENTAURS (2)	154
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 306M
TOTAL PROGRAM COST	$1401 M

5.4 Galilean Satellite Penetrator Network

5.4.1 Science Objectives and Payload

A seismic network deployed by penetrators at one or all of the Galilean satellites would allow detailed investigations of their internal structure to be made and would help determine the degree of tectonic activity at each of these planet-sized worlds. Surface and subsurface composition analyses would also be made by these surface stations. Specific science objectives for a mission of this kind include:

1. Monitor seismic activity of the target satellite to develop a model of its internal structure;
2. Determine subsurface chemical and mineralogical composition;
3. Measure local magnetic field and interaction of satellite surface with the magnetosphere of Jupiter;
4. Determine heat flow properties of the satellite; and
5. Measure the physical properties of the surface material.

The science instruments needed to carry out these objectives for both the carrier spacecraft and penetrators have been listed in Table 5-7. Instrument heritage for these devices is based on similiar instruments used on Galileo and various Earth-orbiting spacecraft. The orbiter/bus carries seven instruments, five of which are dedicated to remote sensing of the satellite's surface. The remaining instruments will be used to monitor Jupiter's magnetosphere and the satellite's interaction with it. Radio Doppler tracking of the orbiter/ bus will make it possible to gather information on the satellite's orbit and large scale features of the satellite (i.e. total mass, diameter, etc.). The total science payload for this vehicle will have a mass of 85 kg. This payload will require a maximum power of 69 W and a data rate in excess of 200,000 bits/sec, if all instruments are operated simultaneously. However, both of these values will be reduced significantly with appropriate scheduling of data gathering and transmission periods. Each penetrator will carry seven instruments to monitor surface and subsurface conditions. The instrument complement has a total mass of 1.9 kg and will require 6.2 W of power if all of the instruments are operating. Data gathered by the penetrator will be transmitted to the orbiter/bus for relay to Earth.

Table 5-7

GALILEAN SATELLITE PENETRATOR NETWORK - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
CCD IMAGER	28	10	100000
MAGNETOMETER	7	4	240
MICROWAVE RADIOMETER	10	10	200
X-RAY SPECTROMETER	11	10	300
UV-VIS-IR MAPPING REFLECTANCE SPECTROMETER	10	10	1-100k
ENERGETIC PARTICLE DETECTOR	9	7	912
RADAR ALTIMETER	10	18	625
TOTAL	85	69	

- **VEHICLE SUBSYSTEMS (EXCLUDING PROPULSION)**

	ORBITER/BUS	PENETRATOR
SCIENCE	85	1.9
STRUCTURE AND DEVICES	190	29.5
THERMAL, CABLING AND PYRO	120	1.0
ATTITUDE AND ARTICULATION CONTROL	81	---
TELECOMMUNICATIONS	20	1.4
ANTENNAS	36	0.6
COMMAND AND DATA	33	1.3
POWER SOURCE AND PROCESSING	144	2.3
LAUNCH TUBE AND RETRO ALLOCATION	---	142
CONTINGENCY (10%)	71	20
	780 kg	200 kg

Table 5-7 (cont'd.)

GALILEAN SATELLITE PENETRATOR NETWORK - PAYLOAD MASS SUMMARY

PENETRATOR

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
3-AXIS SEISMOMETER	0.6	0.09	1.6
α-PROTON BACKSCATTER/X-RAY FLUORESCENCE	0.4	0.1	0.07
TEMPERATURE PROBE	0.07	0.02	2
ACCELERATOR	0.03	0.03	6×10^4 BITS
MAGNETOMETER	0.4	0.07	9.3
FAX IMAGER	0.3	0.9	15
HYDRATED MINERAL ANALYZER	0.15	5.0	VAR.
TOTAL	1.9	6.2	

The orbiter/bus is a 3-axis stabilized vehicle derived from the Mariner Mark II spacecraft. Table 5-7 lists a mass breakdown of the spacecraft by subsystem. For this mission, the spacecraft has a mass of 780 kg exclusive of propellant. The penetrators are based on design studies for application at Mars which were derived from Earth-based vehicles. A summary of the penetrator subsystem masses is also shown in Table 5-7. The orbiter/bus will carry three penetrator vehicles.

5.4.2 Mission Performance

Table 5-8 provides a summary of mission performance data. As with the two previous Jupiter missions, this vehicle will follow a ballistic trajectory from Earth. The spacecraft will be launched in August of 2000 and will require 3.5 years to reach Jupiter. A Ganymede-assisted orbit capture will be effected using an Earth-storable retropropulsion system (I_{sp} = 315 sec). This retro system will require a mass of 2302 kg for both propellants and inerts. The spacecraft will use a satellite touring strategy both to reduce orbit energy (to eventually reach Europa) and to target a penetrator for each of the three outer Galilean satellites. An injected mass of 3866 kg results when all mission hardware elements have been assembled. An on-orbit assembled Centaur(G')/Centaur(G) will be used to escape from Earth orbit. Use of this upper stage results in a 1732 kg mass margin which could be utilized for one or two additional penetrators.

5.4.3 Mission Cost

Estimated costs for this mission are shown in Table 5-9. Spacecraft development, launch, flight operations and data analysis costs will total $868M in FY 1986 dollars. Transportation costs for the Space Shuttle and Centaurs will add $306M which brings the total mission cost to $1174M.

Table 5-8

GALILEAN SATELLITE PENETRATOR NETWORK - PERFORMANCE SUMMARY

- **DESCRIPTION**

FLIGHT MODE	BALLISTIC
TRAJECTORY TYPE	DIRECT
CAPTURE MODE	EARTH-STORABLE RETRO

- **TRAJECTORY**

C_3 $(km/sec)^2$	86.7	LAUNCH DATE	8/5/2000
DLA (deg)	30.0	TOTAL TRIP TIME (yrs) ..	3.5

- **MASS PERFORMANCE** L.V. = OOA CENTAUR(G')/CENTAUR(G)

ORBITER		780 kg
PENETRATORS		600
CHEMICAL RETROPROPULSION		2302
PROPELLANT	1969	
INERTS	333	
INITIAL MASS		3682
L.V. ADAPTER		184
INJECTED MASS REQUIRED		3866
INJECTED MASS CAPABILITY		5598
INJECTED MASS MARGIN		1732

Table 5-9

GALILEAN SATELLITE PENETRATOR NETWORK - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 332M
PENETRATORS	108
RTG's	38
VEHICLE INTEGRATION	19
LAUNCH + 30 DAYS OPERATIONS	31
PROGRAM MANAGEMENT	21
CONTINGENCY (30%)	165
TOTAL DEVELOPMENT	$ 714M
FLIGHT OPERATIONS	124
DATA ANALYSIS	30
TOTAL MISSION OPERATIONS	$154M
SHUTTLE LAUNCHES (2)	142
CENTAURS (2)	154
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 306M
TOTAL PROGRAM COST	$1174M

5.5 Europa Orbiter/Lander

5.5.1 Science Objectives and Payload

Europa possesses one of the smoothest surfaces known in the solar system. This may imply possible volcanic activity or upwelling through surface cracks which has the effect of leveling any surface distortions. An orbiter with a surface lander combination could help resolve the origin of this phenomenon. Specific objectives for this mission include:

- **For the Lander**

1. Investigate the internal properties of Europa via measurement of seismic activity;
2. Obtain surface imagery;
3. Measure local magnetic field strength;
4. Observe interaction between Europa's surface and Jupiter's magnetosphere;
5. Analyze the elemental composition of the surface; and
6. Determine the mineralogical and petrologic characteristics of the surface material.

- **For the Orbiter**

1. Map elemental and mineralogical distribution on the surface;
2. Monitor the interaction of Europa with the Jupiter magnetosphere; and
3. Determine the large-scale thermophysical properties of Europa (i.e. heat flow, temperature conductivity, temperature porosity, etc.).

While this mission profile has Europa as its specific target, missions to Ganymede or Callisto could also be made using much of the same hardware and with many of the same science objectives. Europa was selected since it represented the most difficult case of the outer three Galilean satellites in terms of both performance and radiation constraints.

The orbiter spacecraft will use the same instrument complement as the Galilean Satellite Penetrator Network vehicle but the mission is now focused on a single planet rather than a diverse set of objects. The lander contains seven instruments and a sampler assembly to deliver discrete samples to various direct sensing devices. Details of this science payload are shown in Table 5-10. This instrument complement has a total mass of 57 kg and requires a total of 86 W to support all elements of the science package. Data gathered by these instruments would be transmitted to the orbiter to be relayed back to Earth.

The orbiter is functionally the same as that used for the Galilean Satellite Penetrator Network mission but with slight alterations made to deliver and monitor a surface lander. A mass breakdown for this orbiter is shown in Table 5-10 indicating that the total mass must increase to 849 kg (without propulsion) to support this mission. The lander vehicle is assumed to be a derivative of the Viking lander. A subsystem mass breakdown for this vehicle is also shown in Table 5-10. The 556 kg lander will require a 540 kg propulsion system to effect a safe landing.

5.5.2 Mission Performance

Mission performance data for the Europa Orbiter/Lander mission have been summarized in Table 5-11. This mission uses the same ballistic orbit as the Galilean Satellite Penetrator Network mission (launch in August of 2000 with a 3.5 year flight time). The propulsion system needed to effect a Ganymede-assisted orbit capture at Jupiter and the final orbit insertion at Europa will require a mass of 3884 kg (Earth-storable propellant plus inerts, I_{sp} = 315 sec). The spacecraft orbit will be pumped down to the orbit radius of Europa through the use of successive flybys of other Galilean satellites. When all hardware elements for this mission have been assembled, the spacecraft will have a total mass of 6120 kg. This exceeds the capacity of an on-orbit-assembled Centaur(G')/Centaur(G) for the given trajectory. Thus a higher capacity OTV of the type hypothesized for use at the Space Station must be used for Earth escape. This vehicle provides a positive mass margin of 1848 kg.

Table 5-10

EUROPA ORBITER/LANDER - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**
  (See Galilean Satellite Network for Orbiter Science Payload)

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
FAX IMAGER	7.3	34	4000
3-AXIS SEISMOMETER	2.2	4	1.7
MAGNETOMETER	0.4	0.07	
α- PROTON BACKSCATTER/ X-RAY FLUORESCENCE	2.0	1.5	10
SCANNING ELECTRON MICROSCOPE/ MICROPROBE	25	10	(?)
TEMPERATURE SENSOR	0.1	0.03	500/DAY
PETROGRAPHIC MICROPROBE	5.0	1	2×10^6/IMAGE
SAMPLER ASSEMBLY	15	35.4	---
TOTAL	57.0	86.0	

- **VEHICLE SUBSYSTEMS (EXCLUDING PROPULSION)**

	ORBITER	LANDER
SCIENCE	85	57
STRUCTURE AND DEVICES	200	117
THERMAL, CABLING AND PYRO	87	58
ATTITUDE AND ARTICULATION CONTROL	106	74
RADIATION SHIELDING	40	34
TELECOMMUNICATIONS	61	22
ANTENNAS	8	10
COMMAND AND DATA	44	9
POWER SOURCE AND PROCESSING	141	118
CONTINGENCY (10%)	77	57
TOTAL	849 kg	556 kg

Table 5-11

EUROPA ORBITER/LANDER - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... DIRECT

  CAPTURE MODE .... EARTH-STORABLE RETRO

- **TRAJECTORY**

  C_3 $(km/sec)^2$ .... 86.7   LAUNCH DATE 8/5/2000

  DLA (deg) .... 30.0   TOTAL TRIP TIME (yrs) .. 3.5

- **MASS PERFORMANCE**   L.V. = OTV(4-R)/OTV(2-E)

  ORBITER .... 849 kg
  LANDER 556
  LANDER PROPULSION 540

  CHEMICAL RETROPROPULSION 3884
  PROPELLANT 3366
  INERTS 518

  INITIAL MASS 5829
  L.V. ADAPTER 291

  INJECTED MASS REQUIRED 6120

  INJECTED MASS CAPABILITY 7968
  INJECTED MASS MARGIN 1848

5.5.3 Mission Cost

The estimated cost for this mission is shown in Table 5-12. All mission element costs except transportation require a total of \$1305M in FY 1986 dollars. Transportation costs will add \$161M to this mission scenario bringing the total program cost to \$1466M.

Table 5-12

EUROPA ORBITER/LANDER - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 332M
LANDER VEHICLE	366
RTG's	40
VEHICLE INTEGRATION	32
LAUNCH + 30 DAYS OPERATIONS	49
PROGRAM MANAGEMENT	62
CONTINGENCY (30%)	264
TOTAL DEVELOPMENT	$1145M
FLIGHT OPERATIONS	128
DATA ANALYSIS	32
TOTAL MISSION OPERATIONS	$160M
SHUTTLE LAUNCH	71
OTV'S (2)	80
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 161M
TOTAL PROGRAM COST	$1466M

5.6 Titan Orbiter/Penetrator Network

5.6.1 Science Objectives and Payload

An orbiter spacecraft at Titan will allow long-term observation and analysis of Titan's unusual atmosphere as well as provide information about the nature of its surface. Specific science objectives for this mission include:

1. Conduct radar mapping of Titan's surface to provide image and topographical data;
2. Conduct spectroscopic observations of the atmosphere and thermal radiation emitted by the surface;
3. Conduct radio occultation experiments of the upper atmosphere to obtain temperature and density profiles;
4. Obtain near-IR images of the surface;
5. Determine physical characteristics of Titan via gravity perturbations of the spacecraft's orbit;
6. Characterize Titan's magnetic field and study its interaction with Saturn's magnetosphere; and
7. Perform low orbit aeronomy experiments.

In addition to global investigations carried out from orbit, in situ experiments conducted on the surface are highly desirable. A network of surface penetrators would be useful in monitoring the long-term variations in Titan's weather and as a means of establishing a model of the interior through the use of a seismic network. Specific objectives for these penetrators include:

1. Measure the physical properties of the surface upon impact;
2. Conduct chemical analyses of subsurface material;
3. Determine the subsurface stratigraphy;
4. Conduct long-term weather observations;
5. Measure the local magnetic field strength; and
6. Monitor seismic activity to develop a model of Titan's internal structure.

This mission concept, and the one which follows, have been studied extensively in Reference 4 from which the data presented here have been excerpted. The science payload for both the orbiter and penetrators has been summarized in

Table 5-13. The orbiter carries five direct sensing instruments and four remote sensing devices, mostly of Galileo heritage. The radar mapper, however, will draw on Venus Radar Mapper heritage. The total science package for this vehicle will have a mass of 131 kg with a power requirement of 106 W if all instruments are operating simultaneously. A 3-axis stabilized spacecraft bus will be required to complete the detailed radar mapping phase and to communicate with the surface packages. A subsystem breakdown of this bus, which reflects the Mariner Mark II heritage, can be found in Table 5-13. The surface packages which this bus will be supporting consist of three penetrators of the type described in the Galilean Penetrator Network Mission concept. These vehicles will be placed in diverse locations based on information derived from the radar map.

5.6.2 Mission Performance

This vehicle will fly a direct ballistic trajectory to Titan after launch from Earth in July of 1999. After a flight time of 4.5 years the spacecraft will be placed in orbit around Titan through the use of aerocapture. Further details of this trajectory have been assembled in Table 5-14. The entire spacecraft stack with aerocapture vehicle has a mass of 1960 kg at departure from Earth. A Centaur(G') (fully fueled on orbit) with a Star 48 solid rocket motor will be required to escape from low-Earth orbit. A mass margin of only 45 kg is available with this vehicle indicating that a larger escape stage may be required.

5.6.3 Mission Cost

The estimated cost to carry out this mission has been summarized in Table 5-15. The total project cost including the spacecraft with its aeroshell, launch, flight operations and data analysis is estimated to be $1016M in FY 1986 dollars. Transportation costs will add $231M bringing the total program cost to $1247M.

Table 5-13

TITAN ORBITER/PENETRATOR NETWORK - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
MAGNETOMETER	7	3.7	240
PLASMA WAVE	5	7	500
PLASMA DETECTOR	12	7.2	600
NEUTRAL MASS SPECTROMETER	3.8	12	---
DUST PARTICLE ANALYZER	4.2	1.7	12
UV SPECTROMETER	3	2.5	2500
THERMAL IR SPECTROMETER	18	12	11500
NEAR-IR IMAGER	28	10	---
RADAR MAPPER	50	50	5000
TOTAL	131	106.1	

- **VEHICLE SUBSYSTEM MASSES (EXCLUDING PROPULSION)**

	ORBITER	PENETRATOR
SCIENCE	131 kg	2.1
STRUCTURE AND DEVICES	212	29.5
THERMAL, CABLING AND PYRO	69	1.0
ATTITUDE AND ARTICULATION CONTROL	86	---
TELECOMMUNICATIONS	45	1.4
ANTENNAS	10	0.6
COMMAND AND DATA	33	1.3
POWER SOURCE AND PROCESSING	78	2.3
ENTRY AND DECELERATION	---	51.8
CONTINGENCY (10%)	60	10
TOTAL	739 kg	100 kg

Table 5-13 (cont'd.)

TITAN ORBITER/PENETRATOR NETWORK - PAYLOAD MASS SUMMARY

PENETRATOR

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
3-AXIS SEISMOMETER	0.6	0.09	16
HEAT FLOW SENSORS	0.07	0.02	2
α-PROTON BACKSCATTER/X-RAY FLUORESCENCE	0.4	0.10	0.07
METEOROLOGY	0.3	0.08	5.2
IMPACT ACCELERATOR	0.03	0.07	9.3
FAX IMAGER	0.3	0.90	15
MAGNETOMETER	0.4	0.07	9.3
TOTAL	2.1	1.33	

Table 5-14

TITAN ORBITER/PENETRATOR NETWORK - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... DIRECT

  CAPTURE MODE .... AEROCAPTURE

- **TRAJECTORY**

C_3 (km/sec)^2	115.0	LAUNCH DATE	7/1999
DLA (deg)	-8.0	TOTAL TRIP TIME (yrs)	4.5

- **MASS PERFORMANCE** L.V. = OOA CENTAUR(G')/STAR 48

ORBITER	739 kg
PENETRATOR LAUNCH SYSTEM	45
PENETRATORS (3)	300
AEROCAPTURE VEHICLE	523
CHEMICAL PROPULSION	260
INITIAL MASS	1867
L.V. ADAPTER	93
INJECTED MASS REQUIRED	1960
INJECTED MASS CAPABILITY	2005
INJECTED MASS MARGIN	45

Table 5-15

TITAN ORBITER/PENETRATOR NETWORK - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$248M
PENETRATORS (3)	150
AEROCAPTURE VEHICLE	105
RTG's	27
VEHICLE INTEGRATION	21
LAUNCH + 30 DAYS OPERATIONS	34
PROGRAM MANAGEMENT	46
CONTINGENCY (30%)	189
TOTAL DEVELOPMENT	$820M
FLIGHT OPERATIONS	168
DATA ANALYSIS	28
TOTAL MISSION OPERATIONS	$196M
SHUTTLE LAUNCHES (2)	142
CENTAUR	77
STAR 48	2
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 231M
TOTAL PROGRAM COST	$1247M

5.7 Titan Orbiter/Buoyant Station

5.7.1 Science Objectives and Payload

This mission concept is essentially a growth version of the Titan Orbiter/Penetrator Network discussed in the previous section. The science objectives and instrument complement for the orbiter remain unchanged. However, the three penetrators are now replaced by three small haze probes whose objective will be to collect data regarding the haze layers in Titan's upper atmosphere. The science package for this probe is shown in Table 5-16 and has an allocated mass of 80 kg to support the 16 kg science package through atmospheric entry and data collection.

The second part of this mission is a detailed investigation of the atmosphere and surface using three small balloons and a single buoyant station. The specific science objectives for these four vehicles include:

1. Observe atmospheric circulation and weather conditions on a planet-wide basis;
2. Image and map compositional differences of surface regions;
3. Conduct on-board chemical and elemental analyses of surface samples; and
4. Determine compositional differences, if any, between the northern and southern hemisphere.

The science instruments needed to carry out these objectives are listed in Table 5-16. The buoyant station will require a total payload mass of 80.4 kg and will use a maximum of 144 W to operate these devices. The buoyant station will use an inflatable airship to hold the vehicle at an altitude of 5 km. The airship will be powered and thus can provide directional control which will allow the vehicle to be targeted for a specific site. The small balloons will each carry a science payload of 20 kg and will use no more than 45 W of power. These balloons will be held aloft at an altitude of 50 km and will drift with the local air currents. Both types of vehicles will use conical deceleration modules similar to those used for Pioneer Venus and Galileo. A mass breakdown of vehicle subsystems, including the deceleration module, is shown in Table 5-16. All four probes will be delivered by a spin-stabilized carrier spacecraft, a mass breakdown of which is also shown in Table 5-16.

Table 5-16

TITAN ORBITER/BUOYANT STATION - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD - BUOYANT STATION**

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
NEPHELOMETER	4.4	13.5	10
or			
CLOUD PARTICLE SPECTROMETER	4.4	20	15
GAS CHROMATOGRAPH/MASS SPECTROMETER	15	40	100
ATMOSPHERIC STRUCTURE	4	6	20
LIGHTNING DETECTOR	1	1	10
AEROSOL COLLECTOR	6	10	20
NET FLUX RADIOMETER	3	7	20
RADAR ALTIMETER	10	15	10
NEAR-IR IMAGER	10	10	1000
SURFACE SAMPLER	15	25	---
IR SPECTROMETER	12	10	30
TOTAL	80.4	144 (MAX)	

- **HAZE PROBES AND SMALL BALLOONS**

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)	HAZE PROBE	SMALL BALLOON
GAS CHROMATOGRAPH	2	9	20	X	X
MASS SPECTROMETER	4	13	6	X	
ATMOSPHERE STRUCTURE	4	6	20	X	X
AEROSOL SAMPLE COLLECTOR	6	(10)	--	X	X
LIGHTNING DETECTOR	1	1	10		X
NET FLUX RADIOMETER	3	7	20		X
NEPHELOMETER	4	13	10		X

Table 5-16 (cont'd.)

TITAN ORBITER/BUOYANT STATION - PAYLOAD MASS SUMMARY

- **BUOYANT STATION AND SMALL BALLOON SUBSYSTEMS**

	SMALL BALLOON	BUOYANT STATION
SCIENCE	20	80
STRUCTURE AND DEVICES	28	85
THERMAL CONTROL AND CABLING	12	25
TELECOMMUNICATIONS	11	11
COMMAND AND DATA	14	23
POWER AND HEAT SOURCE	15	102
PROPULSION	---	14
TETHER SYSTEM	---	20
PAYLOAD SUBTOTAL	100 kg	360 kg
H_2 BUOYANT GAS	10	32
RESERVE GAS AND TANK	5	15
BALLOON FABRIC (DIAMETER)	12 (8.2m)	7 (11.9x4m)
FLOATED MASS SUBTOTAL	127 kg	414 kg
GAS TRANSPORT SYSTEM	90	288
AERODECELERATION MODULE	79	219
TOTAL SYSTEM MASS	296 kg	921 kg

- **PROBE CARRIER**

STRUCTURE AND DEVICES	191 kg
THERMAL, CABLING AND PYRO	60
ATTITUDE CONTROL	17
TELECOMMUNICATIONS	21
COMMAND AND DATA HANDLING	30
POWER	94
CONTINGENCY (10%)	41
TOTAL	454 kg

5.7.2 Mission Performance

The orbiter portion of this mission is basically the same as that used for the previous mission. Details are listed in Table 5-17. The probe carrier with the three small balloons and the buoyant station will be launched one year later to arrive after the orbiter is in position to act as a communication relay platform. The four probes will then be targeted for a direct entry while the probe carrier continues on a flyby trajectory. The total injected mass for the probe carrier vehicle stack is shown in Table 5-18. This is 2730 kg heavier than the orbiter stack. As such this vehicle requires a Centaur(G')/Centaur(G), assembled and fueled on orbit, to boost the spacecraft on the proper interplanetary trajectory. The mass margin for this combination of upper stage, spacecraft, and trajectory is a comfortable 970 kg.

5.7.3 Mission Cost

The estimated cost to carry out this mission is shown in Table 5-19. All vehicle hardware, integration, launch, flight operations and data analysis will cost an estimated $2020M in FY 1986 dollars (including a 30% contingency through launch). Transportation costs will add $461M to bring the total program cost to $2481M.

Table 5-17

TITAN BUOYANT STATION (ORBITER) - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... DIRECT

  CAPTURE MODE .... AEROCAPTURE

- **TRAJECTORY**

  C_3 $(km/sec)^2$ .... 115.0   LAUNCH DATE 7/1999

  DLA (deg) .... -8.0   TOTAL TRIP TIME (yrs) .. 4.5

- **MASS PERFORMANCE**   L.V. = OOA CENTAUR(G')/STAR 48

ORBITER	739 kg
PROBE LAUNCH SYSTEM	45
PROBES (3)	240
AEROCAPTURE VEHICLE	523
CHEMICAL PROPULSION	248
INITIAL MASS	1795
L.V. ADAPTER	90
INJECTED MASS REQUIRED	1885
INJECTED MASS CAPABILITY	2005
INJECTED MASS MARGIN	120

Table 5-18

TITAN BUOYANT STATION (PROBE CARRIER) - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... DIRECT

  CAPTURE MODE .... NONE (FLYBY)

- **TRAJECTORY**

  C_3 $(km/sec)^2$ .... 112.0   LAUNCH DATE 7/2000

  DLA (deg) .... -8.2   TOTAL TRIP TIME (yrs) .. 4.2

- **MASS PERFORMANCE**   L.V. = 00A CENTAUR(G')/CENTAUR(G)

CARRIER BUS	454 kg
PROBE LAUNCH SYSTEM	80
LARGE BLIMP	921
SMALL BALLOON PROBES (3)	888
CHEMICAL PROPULSION	257
INITIAL MASS	2600
L.V. ADAPTER	130
INJECTED MASS REQUIRED	2730
INJECTED MASS CAPABILITY	3700
INJECTED MASS MARGIN	970

Table 5-19

TITAN ORBITER/BUOYANT STATION - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 248M
HAZE PROBES	126
AEROCAPTURE VEHICLE	105
CARRIER SPACECRAFT	109
BALLOON PROBES	175
AIRSHIP	227
RTG's	56
VEHICLE INTEGRATION	42
LAUNCH + 30 DAYS OPERATIONS	67
PROGRAM MANAGEMENT	128
CONTINGENCY (30%)	385
TOTAL DEVELOPMENT	$1668M
FLIGHT OPERATIONS	308
DATA ANALYSIS	44
TOTAL MISSION OPERATIONS	$352M
SHUTTLE LAUNCHES (3)	213
CENTAURS (3)	231
STAR 48	2
SPACE-BASED MATING	15
TOTAL TRANSPORTATION	$ 461M
TOTAL PROGRAM COST	$2481M

5.8 Saturn Ring Rover

5.8.1 Science Objectives and Payload

In order to develop an accurate model for the origin and evolution of the Saturn system, a detailed investigation of its ring system is necessary. Specific objectives for a mission of this type (Ref. 5) include:

1. Determine the size, shape, and spatial distribution of the ring particles as well as their dynamical, electrical and optical properties;
2. Determine the ring thickness;
3. Determine the chemical and elemental composition of the ring particles and assess any variations as a function of the distance from Saturn;
4. Observe the rings' interaction with Saturn's magnetosphere and gravity field as well as other effects caused by thermal forcing functions and meteoroid activity; and
5. Observe the interaction of the outer rings with "shepherding moons".

The science payload for this mission (shown in Table 5-20) consists of a wide angle CCD imager, a radar altimeter and eight other remote sensing and direct sensing instruments. Heritage for a majority of these devices is from the Galileo spacecraft. With this instrument complement, the total science payload mass reaches 141 kg. The radar altimeter has a relatively high power usage when compared to that normally expected on an interplanetary spacecraft. This higher power, and the finer resolution it implies, are the result of the nuclear reactor carried on-board this vehicle. The need for a nuclear reactor results from a requirement to use non-Keplerian orbits to essentially hover above the ring plane by less than 100 km, while spiralling in towards the surface of Saturn. This type of trajectory is only possible if continuous thrust is applied at an offset angle to the orbit motion (Ref. 5). A possible configuration for a vehicle to carry out this mission is depicted in Figure 5-1 which would use the SP-100 space nuclear reactor technology (Ref. 17). Specific spacecraft subsystems which are different from the Mariner Mark II heritage assumed for other missions in this study will be needed for use with this low-thrust system. Spacecraft subsystems for this mission, which yield a total spacecraft mass of 1500 kg, are listed in Table 5-20.

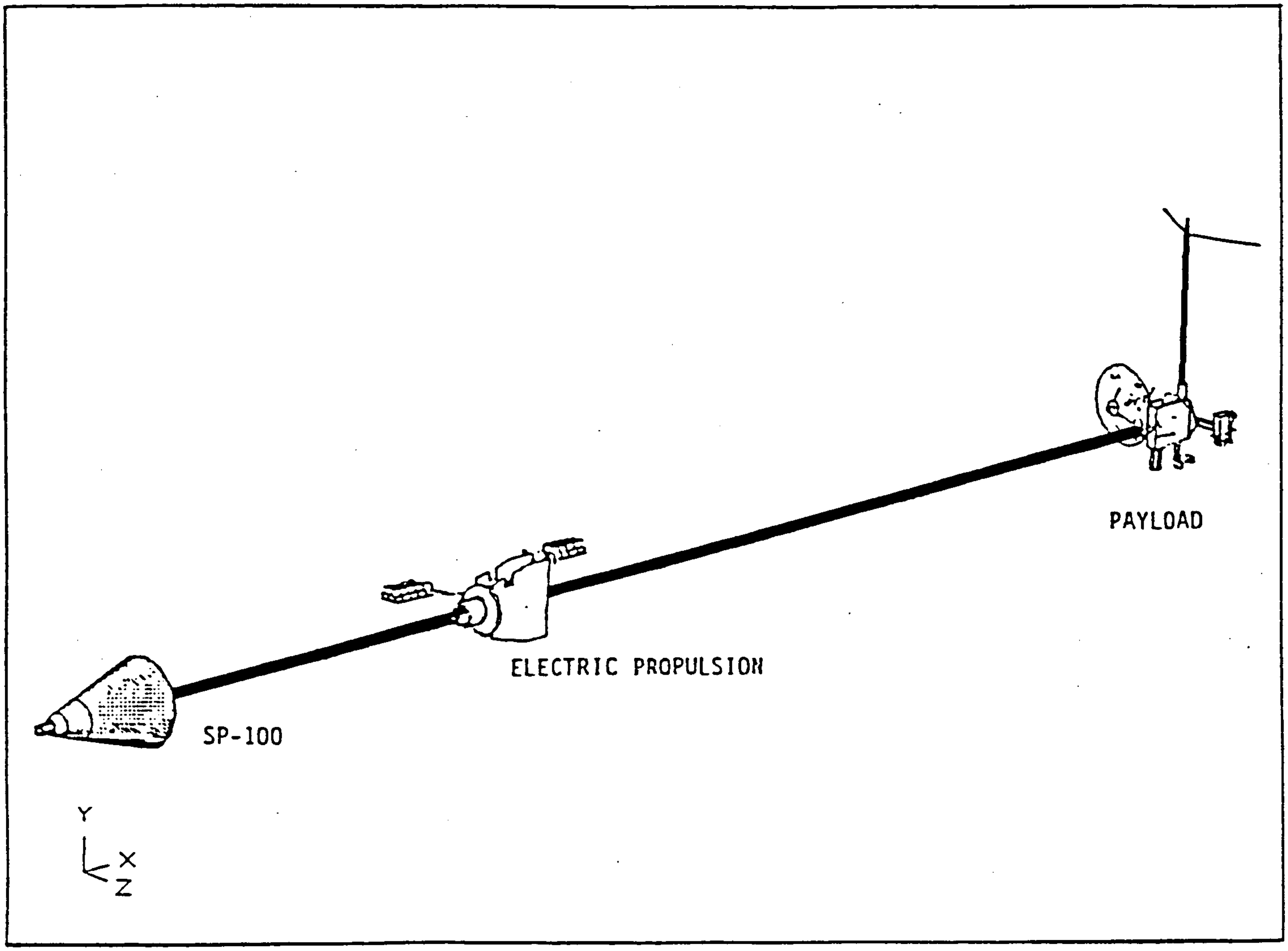

Figure 5-1. Nuclear Electric Propulsion Saturn Ring Rendezvous Spacecraft Configuration

Table 5-20

SATURN RING ROVER - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
CCD IMAGER	28	10	$1x10^6$
MAGNETOMETER	7	4	240
PLASMA DETECTOR	12	7	600
DUST ANALYZER	4	2	12
SCANNING ELECTRON MICROSCOPE PARTICLE ANALYZER	12	10	$5x10^4$
MAPPING REFLECTANCE SPECTROMETER	10	10	$1\text{-}100x10^3$
MICROWAVE RADIOMETER	10	10	200
ION MASS SPECTROMETER	9	11	2000
ENERGETIC PARTICLE DETECTOR	9	7	912
RADAR	40	6000	$5x10^4$
TOTAL	141		

- **ORBITER SUBSYSTEMS (EXCLUDING PROPULSION)**

SCIENCE	141 kg
STRUCTURE AND DEVICES	325
THERMAL, CABLING AND PYRO	111
ATTITUDE AND ARTICULATION CONTROL	180
TELECOMMUNICATIONS	150
ANTENNAS	50
COMMAND AND DATA	48
POWER SUPPORT	200
SUPPORT BOOM	120
CONTINGENCY (10%)	175
TOTAL	1500 kg

5.8.2 Mission Performance

Performance data for this mission have been summarized in Table 5-21. Due to the low-thrust propulsion system, no fixed launch date must be set. The total flight time will be 10.4 years. This flight time includes a spiral escape from Earth in addition to the interplanetary trajectory. The complete flight scenario begins with a Space Shuttle launch of the vehicle stack to a "nuclear-safe" circular orbit of 700 km altitude. The nuclear reactor is then started and the spacecraft spirals away from Earth to follow a direct low-thrust trajectory to Saturn. The spacecraft stack with the NEP stage and load of mercury propellant has a total mass of 16736 kg. The Space Shuttle can deliver 20000 kg to a 700 km circular orbit which yields a 3264 kg mass margin.

5.8.3 Mission Cost

Costs for this mission have been summarized in Table 5-22. The project cost, including spacecraft development, NEP stage acquisition (no development costs), launch and flight operations as well as data analysis are estimated to total $1170M in FY 1986 dollars. Transportation costs will add $75M to bring the total program cost to $1245M.

Table 5-21

SATURN RING ROVER - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... NEP

  TRAJECTORY TYPE .... DIRECT WITH SPIRAL ESCAPE

  CAPTURE MODE .... SPIRAL

- **TRAJECTORY**

  C_3 $(km/sec)^2$ 0

  DLA (deg) .... ANY

  TOTAL TRIP TIME (yrs) 10.4

- **MASS PERFORMANCE** L.V. = SHUTTLE

ORBITER BUS	1500 kg
NEP STAGE (DRY)	2145
SP-100	3000
NEP PROPELLANT	10086
INITIAL MASS	16731
INITIAL MASS CAPABILITY .... OF SHUTTLE @ 700 km	20000
LAUNCH MARGIN	3269

5.9 Uranus Orbiter/Probe

5.9.1 Science Objectives and Payload

The mission described in this section is essentially an uprated version of the flyby/probe concept outlined in the SSEC Core Program. This version will be a Galileo-type mission providing a reconnaissance of the Uranian satellites and ring system as well as characterizing the atmosphere of Uranus. Specific objectives include:

1. Determine the size and structure of the magnetosphere;
2. Determine the nature and composition of the ring system;
3. Determine the composition and geologic history of the satellites;
4. Observe the structure and dynamics of the atmosphere; and
5. Determine the internal structure, composition and dynamics of the atmosphere using an entry probe.

The current science payload which will be used to meet these objectives is listed in Table 5-23. The payload for the orbiter consists of six different instruments. Three of these instruments will investigate particles and fields phenomena; the remainder includes an imaging system, a remote sensing device designed to investigate infrared emissions and a dust detection instrument. This payload has a total mass of 60 kg and will be carried on a Mariner Mark II bus which itself has a mass of 756 kg.

The payload for the probe also consists of six instruments, four of which are remote sensing devices; the remaining two are direct sensing devices. This instrument package has a total mass of 24.4 kg. The remainder of the probe (descent module and entry shield) has a mass of 215.6 kg for a total vehicle mass of 240 kg.

5.9.2 Mission Performance

Two options for delivery of this vehicle to Uranus orbit are summarized in Tables 5-24 and 5-25. Both options assume that the vehicle is placed in an orbit with a three Neptune radii periapse and a 60-day period.

Table 5-22

SATURN RING ROVER - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 429M
NEP STAGE	155*
VEHICLE INTEGRATION	24
LAUNCH + 30 DAYS OPERATIONS	38
PROGRAM MANAGEMENT	29
CONTINGENCY (30%)	203
TOTAL DEVELOPMENT	$878M
FLIGHT OPERATIONS	264
DATA ANALYSIS	28
TOTAL MISSION OPERATIONS	$292M
SHUTTLE LAUNCH	71
ORBITER PROPELLANT AUGMENTATION	4
TOTAL TRANSPORTATION	$ 75M
TOTAL PROGRAM COST	$1245M

*Unit cost only based on prior development for other applications. Pro rata development cost may be required.

Table 5-23

URANUS ORBITER/PROBE - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER	MASS (KG)
CCD IMAGER	21.5
THERMAL IR SPECTRAL RADIOMETER	8.0
DUST DETECTOR	5.0
ENERGETIC PARTICLE DETECTOR	9.0
MAGNETOMETER	6.5
PLASMA ANALYZER	10.0
TOTAL	60.0
PROBE	
ATMOSPHERIC STRUCTURE	3.4
NEPHELOMETER	4.4
HELIUM ABUNDANCE DETECTOR	1.4
NET FLUX RADIOMETER	2.7
NEUTRAL MASS SPECTROMETER	10.0
LIGHTNING AND RADIO DETECTOR/ ENERGETIC PARTICLE INSTRUMENT	2.5
TOTAL	24.4 kg

- **VEHICLE SUBSYSTEMS**

	ORBITER	PROBE
SCIENCE	60	24.4
STRUCTURES AND DEVICES	302	63.3
THERMAL, CABLING AND PYRO	133	25.8
ATTITUDE AND ARTICULATION CONTROL	81	-
TELECOMMUNICATIONS	40	12.9
ANTENNAS	8	(WITH TELECOMMUNICATIONS)
COMMAND AND DATA	24	19.0
POWER SOURCE AND PROCESSING	168	13.6
ENTRY SHELL		65.3
CONTINGENCY	-	15.7
TOTAL	816 kg	240.0 kg

Table 5-24

URANUS ORBITER/PROBE - CHEMICAL RETROPROPULSION OPTION

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... JUPITER SWINGBY

  CAPTURE MODE .... EARTH-STORABLE RETRO

- **TRAJECTORY**

  C_3 (km/sec)^2 ..... 105.1   LAUNCH DATE .... 4 FEB 1995

  DLA (deg.) ...... -19.3   TOTAL TRIP TIME (yrs)..... 11.0

- **MASS PERFORMANCE**   L.V. = OOA CENTAUR(G')/IM

ORBITER		816
PROBE		240
CHEMICAL RETROPROPULSION		797
PROPELLANT	642	
INERTS	155	
INITIAL MASS		1853
L.V. ADAPTER		93
INJECTED MASS REQUIRED		1946
INJECTED MASS CAPABILITY		2351
INJECTED MASS MARGIN		405

Table 5-25

URANUS ORBITER/PROBE - AEROCAPTURE OPTION

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... J/U ΔVEGA

  CAPTURE MODE .... AEROCAPTURE

- **TRAJECTORY**

  C_3 $(km/sec)^2$ ..... 28.5   LAUNCH DATE .... 6 JAN 1993

  DLA (deg.) ...... 6.9   TOTAL TRIP TIME (yrs.) 9.0

- **MASS PERFORMANCE**   L.V. = SHUTTLE/CENTAUR(G')

  ORBITER 816
  PROBE 240

  A-C VEHICLE 1147
  CHEMICAL PROPULSION 2714
  PROPELLANT 2333
  INERTS 381

  INITIAL MASS 4917
  L.V. ADAPTER 246

  INJECTED MASS REQUIRED 5163

  INJECTED MASS CAPABILITY 5900
  INJECTED MASS MARGIN 737

The ballistic option uses a Jupiter swingby trajectory to achieve a reasonable mass performance at launch from Earth. This trajectory requires 11 years to deliver the vehicle to an appropriate approach condition. The probe will then be released before the orbiter is captured. A velocity change of approximately 1500 m/s is required to place this vehicle in its desired orbit. Given these conditions, an injected mass of 1946 kg is required. This can be launched by a fully-fueled Centaur(G') with an injection module and still maintain a margin of 405 kg.

The aerocapture option uses a ΔVEGA-type trajectory with a Jupiter swingby to place this vehicle in the vicinity of Uranus within nine years. A propellant mass of 2330 kg is required for the deep space burn phase of this trajectory. An aerocapture vehicle with a mass of 1147 kg will then be required to place the spacecraft in orbit around Uranus. These two mass allocations contribute to an injected mass requirement of 5163 kg. A single Shuttle launch carrying a partially loaded Centaur(G') will be sufficient to place this vehicle on an escape trajectory due to the low C_3.

5.9.3 Mission Cost

A summary of costs for this mission is presented in Table 5-26. From this table it can be seen that total project costs, for all aspects of the mission except transportation, range from $702M for the ballistic option to $837M for the aerocapture option. Similarly, transportation costs range from $231M for the ballistic mission to $148M for the aerocapture mission. The last column shows the estimated cost for the currently planned flyby mission using the same initial assumptions made for the orbiter. Note that all three spacecraft are estimated to have the same cost. This is due to the fact that all three spacecraft must perform the same basic mission and all three options require a propulsion system with approximately the same capacity. As a result, the cost estimate does not show a difference between the vehicles at this level of analysis. A comparison of all three columns shows that total mission costs increase by no more than 30 percent to achieve a longer, more detailed investigation at Uranus.

Table 5-26

URANUS ORBITER/PROBE - COST ESTIMATE

	Chemical Propulsion Option	Aerocapture Option	Current Flyby Mission
ORBITER SPACECRAFT	$165M	$165M	$165M
PROBE	71	71	71
AEROCAPTURE VEHICLE	---	105	---
RTG'S	25	25	25
VEHICLE INTEGRATION	---	12	---
LAUNCH + 30 DAYS	16	23	16
PROGRAM MANAGEMENT	14	32	14
CONTINGENCY	58	87	58
TOTAL DEVELOPMENT	$349	$520	$349
FLIGHT OPERATIONS	328	292	264
DATA ANALYSIS	25	25	10
TOTAL MISSION OPERATIONS	$353M	$317M	$274M
CENTAUR	77	77	77
STAR 48	2	-	-
SPACE-BASED MATING	10	-	-
SHUTTLE LAUNCHES	142	71	71
TOTAL TRANSPORTATION	$231M	$148M	$148M
TOTAL COST	$933M	$985M	$771M

5.10 Uranus Orbiter/Multiprobe

5.10.1 Science Objectives and Payload

The scope of this mission is similar to the Jupiter Deep Probe/Multiprobe with the exceptions that no deep probe will be used and the smaller probes will reach the 100 bar level instead of the 20 bar level. In addition, the Uranus orbiter will carry science instruments to monitor and investigate the magnetosphere, planetary environment, and satellite and ring systems, as well as relaying probe data to Earth.

Table 5-27 lists the science instrumentation derived from the Galileo orbiter that will be used on the Uranus orbiter. It contains five direct sensing instruments, three remote sensing instruments and an imager. The total science mass for the orbiter is 93.2 kg which uses 61.2 W of power if all instruments operate simultaneously. The science instrumentation for the three probes is exactly the same as that for the Jupiter multiprobe. Table 5-27 lists the masses of the orbiter and probes. The net mass of an orbiter plus three probes (not including propulsion) is 1880 kg.

5.10.2 Mission Performance

Trajectory and mass performance information has been summarized in Table 5-28. The mission will utilize a Jupiter swingby trajectory to reach Uranus six years after a February 2007 launch. After aerocapture by the Uranus atmosphere the spacecraft will enter an elliptical orbit and release probes on subsequent passes. The spacecraft stack has a fully-fueled mass of 4955 kg which will be launched from low-Earth orbit by a Space Station-based OTV.

5.10.3 Mission Cost

The cost breakdown for this mission is shown in Table 5-29. Project cost for hardware development, launch, and flight operations (plus 30% contingency through launch) is $1242M. Transportation costs for the Space Shuttle and OTV's add $161M to the project costs yielding a total mission cost of $1403M.

Table 5-27

URANUS ORBITER/MULTIPROBE - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
CCD IMAGER	28	10	100000
MAGNETOMETER	7	3.7	240
PLASMA DETECTOR	12	7.2	600
PLASMA WAVE ANALYZER	6	3.8	240
ENERGETIC PARTICLE DETECTOR	9	7.4	912
UV SPECTROMETER	4	4.2	1000
PHOTOPOLARIMETER/RADIOMETER	5	7.5	180
NEAR-IR MAPPING SPECTROMETER	18	12	11500
DUST DETECTOR	4.2	5.4	240
TOTAL	93.2	61.2	

- **VEHICLE SUBSYSTEMS**

SUBSYSTEM	MASS (KG) ORBITER/BUS	MULTIPROBE
SCIENCE	93	54.5
STRUCTURE AND DEVICES	198	135
THERMAL, CABLING AND PYRO	132	8.9
ATTITUDE AND ARTICULATION CONTROL	81	---
TELECOMMUNICATIONS	20	12.9
ANTENNAS	36	---
COMMAND AND DATA	33	16.7
POWER SOURCE AND PROCESSING	169	13.1
DECELERATION MODULE	---	106.1
CONTINGENCY (10%)	76	INCLUDED
TOTAL	838	347.2 (per probe)

Table 5-27 (cont'd.)

URANUS ORBITER/MULTIPROBE - PAYLOAD MASS SUMMARY

PROBES INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
ATMOSPHERIC STRUCTURE	3.8	13	18
HELIUM ABUNDANCE DETECTOR	1.4	5.5	4
CLOUD PARTICLE SIZE	4.4	20	30
NET FLUX RADIOMETER	2.7	11.8	32
NEUTRAL MASS SPECTROMETER	12.3	42	32
LIGHTNING AND RADIO DETECTOR/ ENERGETIC PARTICLE DETECTOR	2.5	3.3	8
PRE-ENTRY SCIENCE	23.8	22	36
GAS CHROMATOGRAPHY	3.6	20	50
TOTAL	54.5	137.6	

Table 5-28

URANUS ORBITER/MULTIPROBE - PERFORMANCE SUMMARY

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... JUPITER SWINGBY

  CAPTURE MODE .... AEROCAPTURE

- **TRAJECTORY**

  C_3 $(km/sec)^2$ .... 102.2 LAUNCH DATE 2/14/2007

  DLA (deg) .... -21.1 TOTAL TRIP TIME (yrs) .. 6.0

- **MASS PERFORMANCE** L.V. = OTV(4-R)/OTV(2-E)

  ORBITER .... 838 kg
  URANUS PROBES (3 @ 347.2) 1042

  A-C VEHICLE 2218
  CHEMICAL PROPULSION 622
  PROPELLANT 501
  INERTS 120

  INITIAL MASS 4719
  L.V. ADAPTER 236

  INJECTED MASS REQUIRED 4955

  INJECTED MASS CAPABILITY 6337
  INJECTED MASS MARGIN 1382

Table 5-29

URANUS ORBITER/MULTIPROBE - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$317M
MULTIPROBES	211
AEROCAPTURE VEHICLE	105
RTG's	25
VEHICLE INTEGRATION	23
LAUNCH + 30 DAYS OPERATIONS	37
PROGRAM MANAGEMENT	54
CONTINGENCY (30%)	232
TOTAL DEVELOPMENT	$1004M
FLIGHT OPERATIONS	208
DATA ANALYSIS	30
TOTAL MISSION OPERATIONS	$ 238M
SHUTTLE LAUNCH	71
OTV's (2)	80
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 161M
TOTAL PROGRAM COST	$1403M

5.11 **Neptune Orbiter/Dual Probe**

5.11.1 Science Objectives and Payload

The Neptune Orbiter/Dual Probe spacecraft's first mission objective will be to drop probes into the atmospheres of both Neptune and Triton. The orbiter vehicle will then continue this mission with an extended investigation of the Neptune system. Specific objectives for this mission include:

1. Investigate the details of the magnetosphere around Neptune;
2. Determine the composition, structure and dynamics of Neptune's atmosphere;
3. Determine the nature and extent of the recently discovered ring system (dependent on Voyager and further ground-based observations);
4. Map surface features of the satellites;
5. Determine the nature and composition of Triton's atmosphere;
6. Determine the nature of Triton's surface; and
7. Monitor seismic activity of Triton if the probe accelerometer survives impact.

The planned orbiter science instrumentation uses heritage from the Galileo spacecraft and is shown in Table 5-30. The orbiter contains three remote sensing instruments, five particle and field detectors, and a CCD imager. The total science mass for the orbiter is 93.2 kg which will use a maximum of 61.2 W and transmit 3427 bits/sec of data (not including the near-IR mapping spectrometer data rate of 11,500 bits/sec).

Table 5-30 shows the instrument packages for both the Triton and Neptune probes which are derived from Galileo and the Cassini and Titan exploration missions. The Neptune probe has five direct sensing and sampling instruments and three remote sensing instruments with a total payload mass of 30.7 kg requiring 109 W of power and will generate 154 bits/sec of data if all instruments are operating. The Triton probe has four direct sensing and sampling instruments, one remote sampling instrument, one imager, and a pre-entry science package which contains four direct sensing and sampling instruments for upper atmospheric measurements. This probe has a science package mass of 53.6 kg, requires 129.3 W of power and generates 194 bits/sec of data.

Table 5-30

NEPTUNE ORBITER/DUAL PROBE - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
CCD IMAGER	28	10	15
MAGNETOMETER	7	3.7	240
PLASMA DETECTOR	12	7.2	600
PLASMA WAVE ANALYZER	6	3.8	240
ENERGETIC PARTICLE DETECTOR	9	7.4	912
UV SPECTROMETER	4	4.2	1000
PHOTOPOLARIMETER/RADIOMETER	5	7.5	180
NEAR-IR MAPPING SPECTROMETER	18	12	11500
DUST DETECTOR	4.2	5.4	240
TOTAL	93.2	61.2	

PROBES

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)	TRITON PROBE	NEPTUNE PROBE
NEPHELOMETER	4.4	13.5	10	X	X
NEUTRAL MASS SPECTROMETER	12.3	42	32	X	X
ATMOSPHERIC STRUCTURE INSTRUMENT	3.8	13	18	X	X
NET FLUX RADIOMETER	2.7	11.8	32	X	X
DESCENT IMAGER	3	7	16	X	
GAS CHROMATOGRAPH	3.6	20	50	X	X
PRE-ENTRY SCIENCE	23.8	22	36	X	
HELIUM ABUNDANCE DETECTOR	1.4	5.5	4		X
LIGHTNING & RADIO DETECTOR/ ENERGETIC PARTICLE INSTRUMENT	2.5	3.3	8		X

Table 5-30 (cont'd.)

NEPTUNE ORBITER/DUAL PROBE - PAYLOAD MASS SUMMARY

- **ENTRY PROBES**

SUBSYSTEM	TRITON PROBE	NEPTUNE PROBE
SCIENCE	53.6 kg	30.7 kg
STRUCTURE (DESCENT MODULE) AND DEVICES	49.6	142.7
THERMAL, CABLING AND PYRO	8.9	8.9
ATTITUDE AND ARTICULATION CONTROL	---	---
TELECOMMUNICATIONS	12.9	12.9
ANTENNAS	---	---
COMMAND AND DATA	16.7	16.7
POWER	13.1	13.1
DECELERATION MODULE	64.9	106.
ADAPTER	11.3	28
TOTAL	231 kg	359 kg

- **ORBITER SUBSYSTEMS (EXCLUDING PROPULSION)**

	AEROCAPTURE	NEP
SCIENCE	93 kg	93 kg
STRUCTURE AND DEVICES	198	325
THERMAL, CABLING AND PYRO	132	111
ATTITUDE AND ARTICULATION CONTROL	81	180
TELECOMMUNICATIONS	20	150
ANTENNAS	36	50
COMMAND AND DATA	33	48
POWER SOURCE AND PROCESSING	169	200
SUPPORT BOOM	---	120
CONTINGENCY (10%)	76	175
TOTAL	838 kg	1452 kg

Table 5-30 lists subsystem masses for the two probes and two different types of orbiters. The orbiter subsystems will be partially defined by the type of transfer mode used to reach Neptune. Two trajectory options are currently envisioned: low-thrust using nuclear electric propulsion (NEP), and a ballistic Jupiter swingby. For a NEP transfer the total mass of the orbiter and two probes is 2042 kg. For the ballistic Jupiter swingby transfer the total mass is 1467 kg.

5.11.2 Mission Performance

Trajectory and mass performance information for the two transfer modes is given in Tables 5-31 and 5-32. If the NEP mode is chosen the spacecraft will travel directly from Earth to Neptune after being launched from low-Earth orbit by a Space Station-based OTV. Upon arrival at Neptune 8.3 years later, the spacecraft will release a probe to Triton as it is passed on the capture spiral. After subsequent release of the Neptune probe, the orbiter will transfer probe data back to Earth and begin its own scientific observations. The entire spacecraft stack will have a mass of 17649 kg when launched from Earth.

A slightly faster alternative to the NEP flight mode is the Jupiter swingby, ballistic transfer. In this scenario the spacecraft is launched in January of 2006 from low-Earth orbit by a Space Station-based OTV. After a gravity-assist swingby of Jupiter the spacecraft is captured into Neptune orbit by utilizing Neptune's atmosphere for aerocapture. Both probes will be released after aerocapture. For this flight mode the spacecraft stack will have a mass of 3986 kg at Earth departure.

5.11.3 Mission Cost

The estimated costs to carry out both versions of this mission are listed in Table 5-33. For the Jupiter swingby ballistic mode the total mission cost is $1562M while for the NEP mode the cost is $1799M.

Table 5-31

NEPTUNE ORBITER/DUAL PROBE - PERFORMANCE SUMMARY (NEP)

- **DESCRIPTION**

FLIGHT MODE	NEP
TRAJECTORY TYPE	DIRECT
CAPTURE MODE	SPIRAL CAPTURE

- **TRAJECTORY**

C_3 $(km/sec)^2$	36.0
DLA (deg)	-14.2
TOTAL TRIP TIME (yrs)	8.3

- **MASS PERFORMANCE** L.V. = OTV(4-R)/OTV(2-E)

ORBITER	1452 kg
TRITON PROBE	231
NEPTUNE PROBE	359
NEP STAGE (DRY)	2145
SP-100	3000
NEP PROPELLANT	9621
INITIAL MASS	16808
L.V. ADAPTER	841
INJECTED MASS REQUIRED	17649
INJECTED MASS CAPABILITY	18812
INJECTED MASS MARGIN	1163

Table 5-32

NEPTUNE ORBITER/DUAL PROBE - PERFORMANCE SUMMARY (BALLISTIC)

- **DESCRIPTION**

  FLIGHT MODE .... BALLISTIC

  TRAJECTORY TYPE .... JUPITER SWINGBY

  CAPTURE MODE .... AEROCAPTURE

- **TRAJECTORY**

C_3 (km/sec)^2	134.1	LAUNCH DATE	1/18/2006
DLA (deg)	-7.8	TOTAL TRIP TIME (yrs) ...	7.0

- **MASS PERFORMANCE** L.V. = OTV(4-R)/OTV(2-E)

ORBITER		838 kg
TRITON PROBE		231
NEPTUNE PROBE		359
A-C VEHICLE		1868
CHEMICAL PROPULSION		461
PROPELLANT	368	
INERTS	93	
INITIAL MASS		3757
L.V. ADAPTER		190
INJECTED MASS REQUIRED		3947
INJECTED MASS CAPABILITY		4091
INJECTED MASS MARGIN		144

Table 5-33

NEPTUNE ORBITER/DUAL PROBE - COST ESTIMATE

(Costs in FY'86 Dollars)

	JUPITER SWINGBY	NEP
ORBITER SPACECRAFT	$ 316M	$ 429M
NEPTUNE PROBE	149	149
TRITON PROBE	124	124
AEROCAPTURE VEHICLE	105	---
NEP STAGE	---	155*
RTG's	25	---
VEHICLE INTEGRATION	32	39
LAUNCH + 30 DAYS OPERATIONS	49	56
PROGRAM MANAGEMENT	65	74
CONTINGENCY (30%)	260	308
TOTAL DEVELOPMENT	$1125M	$1334M
FLIGHT OPERATIONS	244	272
DATA ANALYSIS	32	32
TOTAL MISSION OPERATIONS	$ 276M	$ 304M
SHUTTLE LAUNCH	71	71
OTV's (2)	80	80
SPACE-BASED MATING	10	10
TOTAL TRANSPORTATION	$ 161M	$ 161M
TOTAL PROGRAM COST	$1562M	$1799M

*Unit cost only based on prior development for other applications. Pro rata development cost may be required.

5.12 **Pluto Orbiter/Lander and Charon Lander**

5.12.1 Science Objectives and Payload

This mission will characterize the Pluto/Charon system as well as measure the farthest known boundary of the interplanetary environment. Specific science objectives will include:

1. Determine the presence of a magnetic field about Pluto;
2. Conduct general planetology investigations of both bodies;
3. Measure the interplanetary environment at Pluto's orbit;
4. Determine the surface conditions and composition of both Pluto and Charon by utilizing hard landers; and
5. Determine the interaction between Pluto and Charon (i.e. tidal effects).

Candidate science instrument packages for the orbiter and landers are listed in Table 5-34. Instrument heritage for the orbiter is based on Galileo and Lunar Polar Orbiter instrument packages. The orbiter science payload consists of 10 instruments of which five are particle and field detectors, four are remote sensing instruments and one is an imager. The total orbiter science payload has a mass of 105.4 kg, requires 77.3 W of power and generates about 900,000 bits/sec of data if all instruments are operating simultaneously. Identical hard landers deployed to the surfaces of Pluto and Charon have science packages based on Viking lander and Mars Penetrator heritages. The lander science payloads consist of three direct sensing and sampling instruments, a magnetometer, and a facsimile camera capable of producing 360° x 90°, 3-color images of the surface. Each lander science package has a mass of 5 kg, requires 7.5 W of power and will produce a data rate of about 28 bits/sec if all instruments are operating. Masses for each of the orbiter and lander subsystems are shown in Table 5-34. The total mass (excluding propulsion) of the vehicle would then be 1585.5 kg.

5.12.2 Mission Performance

A summary of mission performance data is given in Table 5-35. A Space Station-based OTV will be used to give the spacecraft the necessary ΔV to escape Earth and initiate its interplanetary trajectory. The spacecraft will

Table 5-34

PLUTO ORBITER/LANDER AND CHARON LANDER - PAYLOAD MASS SUMMARY

- **SCIENCE PAYLOAD**

ORBITER

INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
CCD IMAGER	28	10	100000
MAGNETOMETER	7	3.7	240
PLASMA DETECTOR	12	7.2	600
ENERGETIC PARTICLE DETECTOR	9	7.4	912
UV-VIS-IR MAPPING REFLECTANCE SPECTROMETER	10	10	1-100k
GAMMA-RAY SPECTROMETER	12	6	2000
RADAR ALTIMETER	7	17.5	625
MULTICHANNEL MICROWAVE RADIOMETER	10	10	200
PLASMA WAVE INSTRUMENT	6	3.8	768k
DUST DETECTOR	4	1.7	12
TOTAL	105	77.3	

- **VEHICLE SUBSYSTEMS (EXCLUDING PROPULSION)**

	ORBITER/BUS	LANDER
SCIENCE	105	5.0
STRUCTURE AND DEVICES	325	11.0
THERMAL, CABLING AND PYRO	111	4.1
ATTITUDE AND ARTICULATION CONTROL	180	3
TELECOMMUNICATIONS	150	2
ANTENNAS	50	0.5
COMMAND AND DATA	48	2
POWER SOURCE AND PROCESSING	200	6
SUPPORT BOOM	120	---
CONTINGENCY	180	3
TOTAL	1469 kg	36.6 kg

Table 5-34 (cont'd.)

PLUTO ORBITER/LANDER AND CHARON LANDER- PAYLOAD MASS SUMMARY

LANDER INSTRUMENT	MASS (kg)	POWER (W)	DATA (BITS/SEC)
3-AXIS SEISMOMETER	2.2	4.0	1.7
FACSIMILE CAMERA	0.3	0.9	15.0
α-PROTON BACKSCATTER/ X-RAY FLUORESCENCE	2.0	1.5	1.0
MAGNETOMETER	0.4	0.07	10.0
TEMPERATURE SENSOR	0.1	1.0	500/day
TOTAL	5.0	7.47	

Table 5-35

PLUTO ORBITER/LANDER AND CHARON LANDER - PERFORMANCE SUMMARY

- **DESCRIPTION**

FLIGHT MODE .... NEP

TRAJECTORY TYPE .... DIRECT

CAPTURE MODE .... SPIRAL CAPTURE TO CHARON'S ORBITAL RADIUS

- **TRAJECTORY**

C_3 $(km/sec)^2$ 49.0

DLA (deg) 0.02

TOTAL TRIP TIME (yrs) 7.7

- **MASS PERFORMANCE** L.V. = OTV(4-R)/OTV(2-E)

ORBITER	1469 kg
CHARON LANDER	53
PLUTO LANDER	63
NEP STAGE (DRY)	2145
SP-100	3000
NEP PROPELLANT	7220
INITIAL MASS	13950
L.V. ADAPTER	698
INJECTED MASS REQUIRED	14648
INJECTED MASS CAPABILITY	15456
INJECTED MASS MARGIN	808

then utilize nuclear electric propulsion (NEP) for interplanetary flight and a spiral capture at Pluto. One lander will be released at Charon as the spacecraft passes it and one lander will be deployed at Pluto as the orbiter reaches the desired final orbit. Low-resolution imaging and mapping will be used to select suitable landing sites.

For a NEP stage dry mass of 2145 kg the total required injected mass is 14648 kg, 808 kg below the OTV's mass capability for a launch to a C_3 of 49 (km/sec).

5.12.3 Mission Cost

A cost estimate breakdown is given in Table 5-36. Total cost for hardware development, launch, flight operations and data analysis is $1328M including 30% contingency through vehicle launch. Total transportation costs for the Shuttle and OTV are $161M giving a total estimated program cost of $1489M.

Table 5-36

PLUTO ORBITER/LANDER AND CHARON LANDER - COST ESTIMATE

(Costs in FY'86 Dollars)

ORBITER SPACECRAFT	$ 429M
LANDERS (2)	85
NEP STAGE	155
RTG's	10
VEHICLE INTEGRATION	28
LAUNCH + 30 DAYS OPERATIONS	43
PROGRAM MANAGEMENT	44
CONTINGENCY (30%)	238
TOTAL DEVELOPMENT	$1032M
FLIGHT OPERATIONS	264
DATA ANALYSIS	32
TOTAL MISSION OPERATIONS	$ 296M
SHUTTLE LAUNCH	71
OTV's (2)	80
SPACE-BASED MATING	10
TOTAL TRANSPORTATION	$ 161M
TOTAL PROGRAM COST	$1489M

*Unit cost only based on prior development for other applications. Pro rata development cost may be required.

6. SUMMARY AND CONCLUSIONS

It is abundantly clear that the outer planets and their satellites offer a diverse and exciting collection of targets for exploration. From the 11 candidate missions described in this report, it can be seen that exploration of the outer solar system can reveal many scientifically interesting details about the origin and evolution of the entire solar system. As technological advances in spacecraft and related systems continue to be made, greater possibilities for exploration will become available.

The Committee on Planetary and Lunar Exploration (COMPLEX) of the Space Science Board has established a number of scientific objectives to increase knowledge and understanding of the planets and moons of the outer solar system. The Solar System Exploration Committee (SSEC), acting on recommendations made by COMPLEX, has established a Core Program of economically and scientifically feasible missions to the inner and outer solar system through the year 2000. The Saturn Orbiter/Titan Probe mission (now renamed Cassini) and the Saturn Flyby/Saturn Probe mission of the Core Program will meet several of the objectives set out by COMPLEX for exploration of the outer solar system. However, many more missions will not be possible, either within the time frame of the Core Program, or because of technological constraints. Therefore, the SSEC has recommended that the Core Program be augmented with technologically challenging missions as soon as national priorities permit.

The purpose of this study has been to examine several possible mission concepts of this type which would be carried out in the 1995-2015 time period. These missions would be more technologically challenging and would fulfill objectives not addressed by the Core Program. Eleven candidate missions were identified and examined in detail. For each mission identified, the following subjects have been addressed:

- Science objectives and payload
- Instruments and expected results
- A brief mission scenario
- Hardware requirements and technological readiness
- Assembled spacecraft mass
- Estimated cost.

The mission set assembled for this study includes four missions to Jupiter and the Galilean satellites, two missions to Titan, and one mission each to Saturn, Uranus, Neptune and Pluto. The final mission in this set examines the consequences of modifying the Uranus Flyby/Probe spacecraft (a currently identified Core Program mission) into an orbiter/probe vehicle. Characteristics of these candidate missions are described in detail in this report.

It is hoped that the realization of the missions in this set will advance the levels of investigation and exploration as defined by COMPLEX of a large portion of the outer solar system. Figure 6-1 illustrates the status of each target currently, after the Core Program is complete, and after the candidate missions in this study have been flown. As this figure demonstrates, past missions to Jupiter and the Galilean satellites have brought our knowledge of these bodies to the exploration phase, with no Core Program missions to either target. The four candidate missions to Jupiter and its moons would bring these targets to the intensive study phase of investigation. Saturn and Titan are currently in the early exploration stage. Proposed Core Program missions, when completed, would bring both targets through the exploration phase, and the mission proposed in this study would bring Saturn and Titan to the intensive study phase of investigation. At the farther reaches of the solar system, Uranus and Neptune are currently at the reconnaissance level of investigation. Core Program missions would enhance information about these planets to the initial stages of exploration, and the two missions proposed in this study would advance further into this phase. Pluto is currently in what could be termed the pre-reconnaissance phase, with the extended Core Program mission to Pluto achieving the reconnaissance phase. The mission proposed for Pluto in this study would bring both Pluto and its satellite, Charon, to a phase of investigation slightly beyond the reconnaissance level.

In the course of the study described in this report, it was found that existing hardware elements and concepts could be used to carry out most of these missions. There were several exceptions which occurred due to the nature of the mission involved. Some of these required unique solutions such as the airship concept used at Titan. Others required hardware, such as NEP

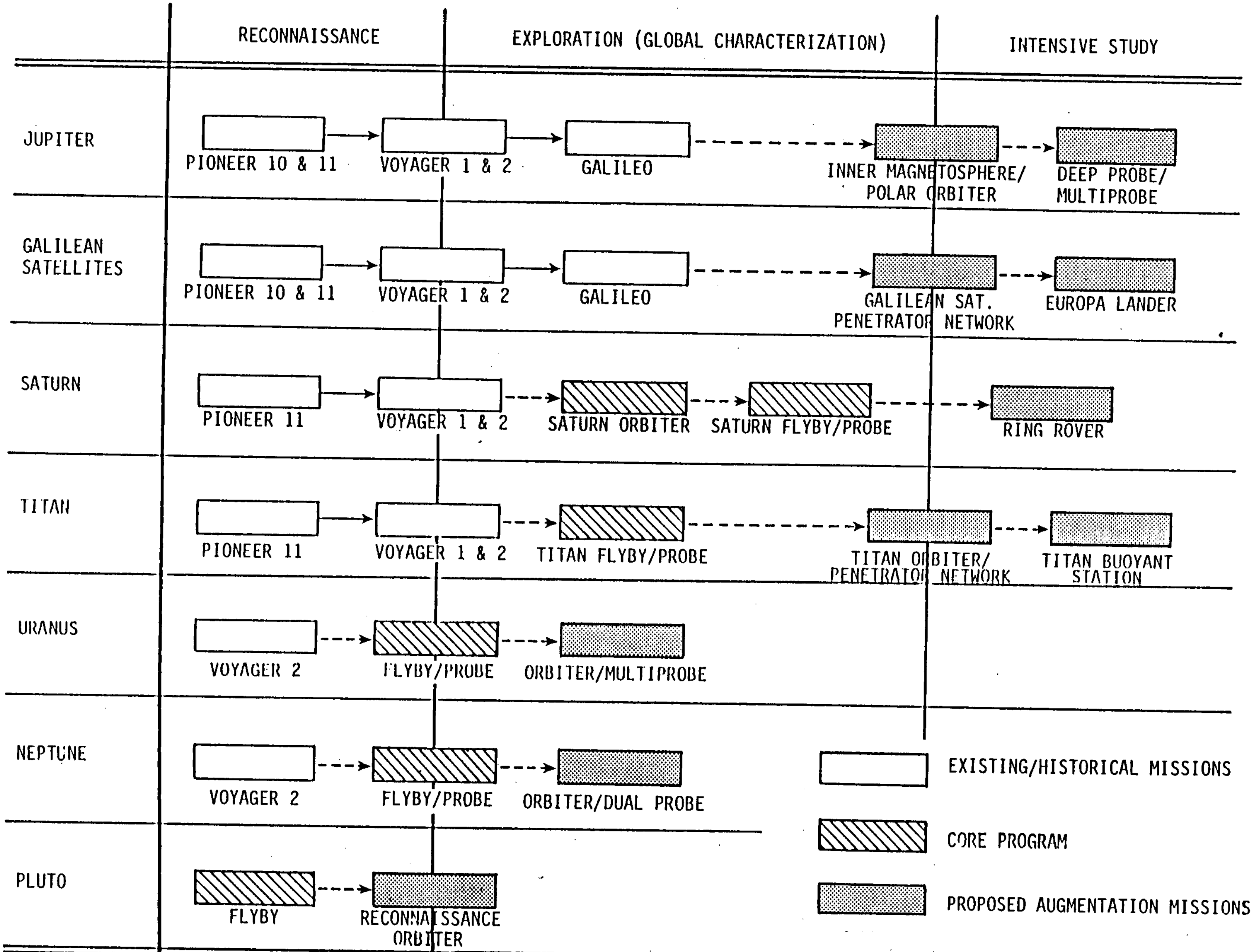

Figure 6-1. Status of Outer Planet Exploration After Proposed Augmentation Missions

or aerocapture, to fulfill mission needs but which would have a wider range of application in the future. The cost for these missions was found to range from $561M to $2286M in FY 1986 dollars. All of the mission concepts tend to be ambitious in scope; in some cases they could be split among international partners reducing the cost for each participating country. Some of the required mission technologies, e.g., aerocapture, nuclear electric propulsion, and high-level automation are indeed challenging from today's perspective, but none should be insurmountable on the time scale ($\sim$2000) suggested for the proposed concepts.

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